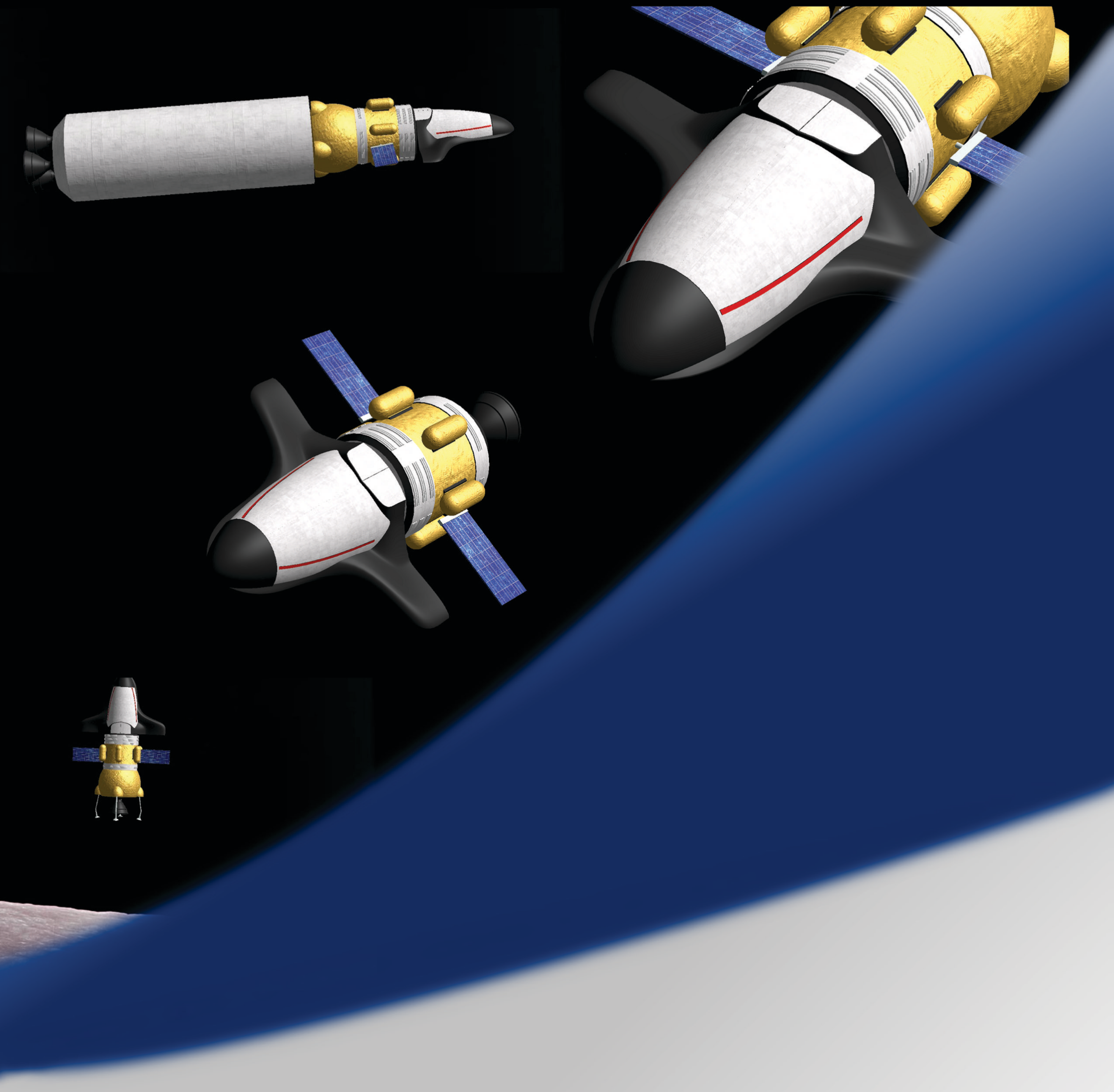
Response to 2005/2006
AIAA FOUNDATION Undergraduate Team
Space Transportation Design Competition



Project Artemis
Presented by Georgia Institute of Technology, Aerospace Engineering Team 2006

Project Artemis

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June 2006

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1 Executive Summary

> "This cause of exploration and discovery is not an option we choose; it is a desire written in the human heart... America will make those words come true."
> — George W. Bush

Quoted in "The Vision for Space Exploration" (VSE), President Bush's words ring true for mankind's insatiable appetite to pioneer new horizons which can only be slowed down by the inability to afford the costs or gain safe access to that horizon. VSE presents such a plan for future human space travel. In conjunction with VSE, the American Institute of Aeronautics and Astronautics (AIAA) has presented a Request for Proposal (RFP) to design a Lunar Transportation System (LTS) in order to safely provide a cost effective means of accessing the moon's existing horizons. Team LETO proudly presents Project Artemis, a safe and reliable solution to the RFP while keeping affordable costs throughout its lifetime. For these reasons, Project Artemis provides the "best value" for AIAA's RFP.

Project Artemis, at a wet mass of 72 mT, transports 4 crewmembers and 500 kg of payload from LEO to the moon, where the crew performs anywhere from 4–7 days of EVAs and returns to a predetermined site on Earth with 100 kg of cargo. At least one such mission is performed every year for Project Artemis's lifetime. To accomplish these tasks at their "best value," Team LETO extensively used system engineering tools such as an AHP, a morphological matrix, and a DSM. However, before using these tools, Team LETO researched the capabilities of the Apollo mission's subsystems in order to gain an understanding of the daunting task to transport life to and from the moon. Team LETO also explored the Exploration Systems Architecture Study (ESAS) to read and evaluate current approaches to VSE.

By incorporating this research with the aforementioned system engineering tools, Team LETO designed Project Artemis with a safe and reliable Earth Orbit Rendezvous (EOR)-Direct mission architecture including an Apollo 13 style "lifeboat." The main benefit of having EOR-Direct architecture is that there will be no Lunar Orbit Rendezvous (LORs) which present a possible dangerous scenario to the crew each time they are performed. Likewise dissimilar to the Apollo program is the makeup of the CM with a use of a Russian Klipper fixed-winged body instead of a capsule. While its safety is comparable to a capsular CM, the fixed-winged body has greater precision and therefore more reliable when it comes to landing at a predetermined site. Like the CM, the SM, though cylindrical like the Apollo's, also shares more similarity to the Klipper's design. It is fitted with two solar panels which provide the power, the habitat environment for the mission, as well as an airlock ensuring a safer and more reliable environment for the crew to live compared with that of the Apollo mission. The LM is very similar to the 1970's version though it only retains Apollo's descent engines as ascent and living space is handled in the SM.

The available subsystems of Project Artemis utilize new technology with moderate-to-high TRLs while making the overall system safer and more reliable than the standard system. The power subsystem uses lightweight UTJ GaAS Solar Panels and Li-Ion Batteries to provide reliable power throughout the mission duration. EDL is handled via the winged body of the CM as well as a parafoil which reduces the mass of the CM and also provides reliable maneuverability. Project Artemis's propulsion uses the extensively handled and readily available NTO/MMH once in LLO. In order to provide the needed interspace propulsion and launch to LEO, Team LETO uses the space shuttle CaLV which is the safest human LV known to date.

For future missions needs, Project Artemis also included the following margins while designing these subsystems: 20% for power, 30% for dry mass, 2% for propulsion, 40% for cost, and 40% for mission schedule. With these margins, Project Artemis uses 6000 W of power and the cost of the entire program is $90B.

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Nomenclature

APS	Auxiliary Propulsion System
C&DH	Command and Data Handling
CaLV	Cargo Launch Vehicle
CEV	Crew Exploration Vehicle
DSN	Deep Space Network
EDS	Earth Departure Stage
EELV	Evolved Expendable Launch Vehicle
ELV	Expendable Launch Vehicle
EVA	Extra-Vehicular Activity
ESAS	Exploration Systems Architecture Study
IVHM	Integrated Vehicle Health Monitoring
LEO	Low Earth Orbit
LETO	Lunar Exploration and Transportation Organization
LH2	Liquid Hydrogen
LLO	Low Lunar Orbit
LM	Lunar Module
LOC	Loss of Crew
LOI	Lunar Orbit Insertion
LOM	Loss of Mission
LOX	Liquid Oxygen
LSAM	Lunar Surface Access Module
LTS	Lunar Transport System
LTV	Lunar Transfer Vehicle
LV	Launch Vehicle
NASA	National Aeronautics and Space Administration
OMS	Orbital Maneuvering System
PMAD	Power Management and Distribution
PMC	Propulsion Mass Calculator
RCS	Reaction Control System
RFP	Request for Proposal
RSRB	Reusable Solid Rocket Booster
RSRM	Reusable Solid Rocket Motor
SSME	Space Shuttle Main Engine
TT&C	Telemetry, Tracking and Command
TDRSS	Tracking and Data Relay Satellite System
TLI	Trans Lunar Insertion
TRL	Technology Readiness Level
T/W	Thrust-to-Weight
UTJ	Ultra-Triple Junction
WCS	Waste Collection System

2 Understanding RFP Requirements

2.1 Project Objective

Project Artemis meets the AIAA RFP requirements of sending humans back to the lunar surface by 2020. Artemis places the safety of the crew above all other considerations.

Team LETO (Lunar Exploration & Transportation organization) has completely read and understood the AIAA RFP and has answered it with a proposal that is unique, safe, reliable and reusable. The spacecraft carries a crew of 4 and will hold up to 500 kg of payload. The lunar operations phase will last up to seven days with the entire crew having the capability to do extravehicular activities (EVA). The system will also be able to support one mission per year.

The safety of the crew was considered the most important factor. In light of recent failures with the safety of the shuttle, Team LETO searched for the optimized solution that takes the well-being of its astronauts. Team LETO is proud to assert that no crew has ever felt safer in any design, as in the Artemis design.

2.2 Design Requirements & Constraints

Project Artemis' design takes into account the requirements and constraints placed by the RFP & the Vision for Space Exploration to ensure that the LTS is manned rated by 2018.

The Vision for Space Exploration requires the humans to land on the moon no later than 2018. This requirement directly affects the design selections that could be made by team LETO by restricting the subsystem to be of a TRL 6 or higher.

The major design constraint for Project Artemis was development and flight readiness of the launch vehicles capable of lifting the crew and the cargo. Apollo performed this task using a $500 million Saturn V rocket. Team LETO considers a shuttle derived launch vehicle for crew and cargo launch vehicle.

The RFP requires the vehicle to be on the surface of the moon for 7 days. This was one of the major design drivers for the power subsystem and the lunar module hatchway design.

Other requirements set forth by the RFP and addressed in the later sections of the proposal in detail includes landing at a predetermined site after earth reentry and reusing the system extensively. Life support system capable of providing life support for a minimum of 7 days on the surface of the moon is required according to the RFP.

3 Baseline Design

3.1 Assumptions

Team LETO made very conservative assumptions to reduce the cost and increase the safety of the mission significantly while complying with all the requirements set forth by the RFP and Vision for Space Exploration.

One key assumption made for the design of Project Artemis is that the mission will not allow for landing on the dark side of the moon. This assumption allows the vehicle to use solar energy as its source of power. It will also decrease the overall complexity of the mission by allowing the system to depend only on one source of energy. This assumption imposes an additional constraint of black out dates depending on the landing location.

The spacecraft will be able to land at any site on the moon including the far side. Landing on the far side requires a communication architecture to be in place, which could then be used to establish a continous communication link between the ground station and the vehicle. Currently, no such communication architecture is available or planned. As a result, even though being able to land on the far side Artemis will not land on the far side until a communication architecture has been placed on moon by NASA.

3.2 Architecture

3.2.1 Mission Architecture

Project Artemis' mission architecture is designed to be a safe and reliable method to get astronauts to and from the moon. The architecture also meets the RFP requirements for originality and sound engineering.

Team LETO's main concern in returning humans to the moon is safety. Due to recent complications with human spaceflight, the need for safety has increased greatly. Consequently, Team LETO decided that stricter measures should be taken to insure safety and reliability. This train of thought can be seen at the top levels of the Artemis project.

The architecture that was decided upon was an Earth Orbit Rendezvous – Direct (EOR-Direct) mission.

Phase 1: The architecture consists of two different launches, which include the spacecraft and two Earth Departure Stages (EDS).

Phase 2: The spacecraft and the Lunar Orbit Insertion (LOI) EDS stage dock with the Lunar Transfer Insertion (LTI) EDS stage, which is placed in orbit 45 days prior to the launch of crew.

Phase 3: At the predetermined spot for LTI, the LTI EDS stage will fire and insert both the spacecraft and the LLO EDS stage into a trajectory towards the moon. After engine shutdown, the LTI EDS stage will be expended.

Phase 4: After a 3-day trip the LOI EDS ignites and ends with the spacecraft entering Low Lunar Orbit (LLO). Once at LLO the spacecraft jettisons the LOI EDS and orbits until all systems are checked out and then begins the descent to the landing site.

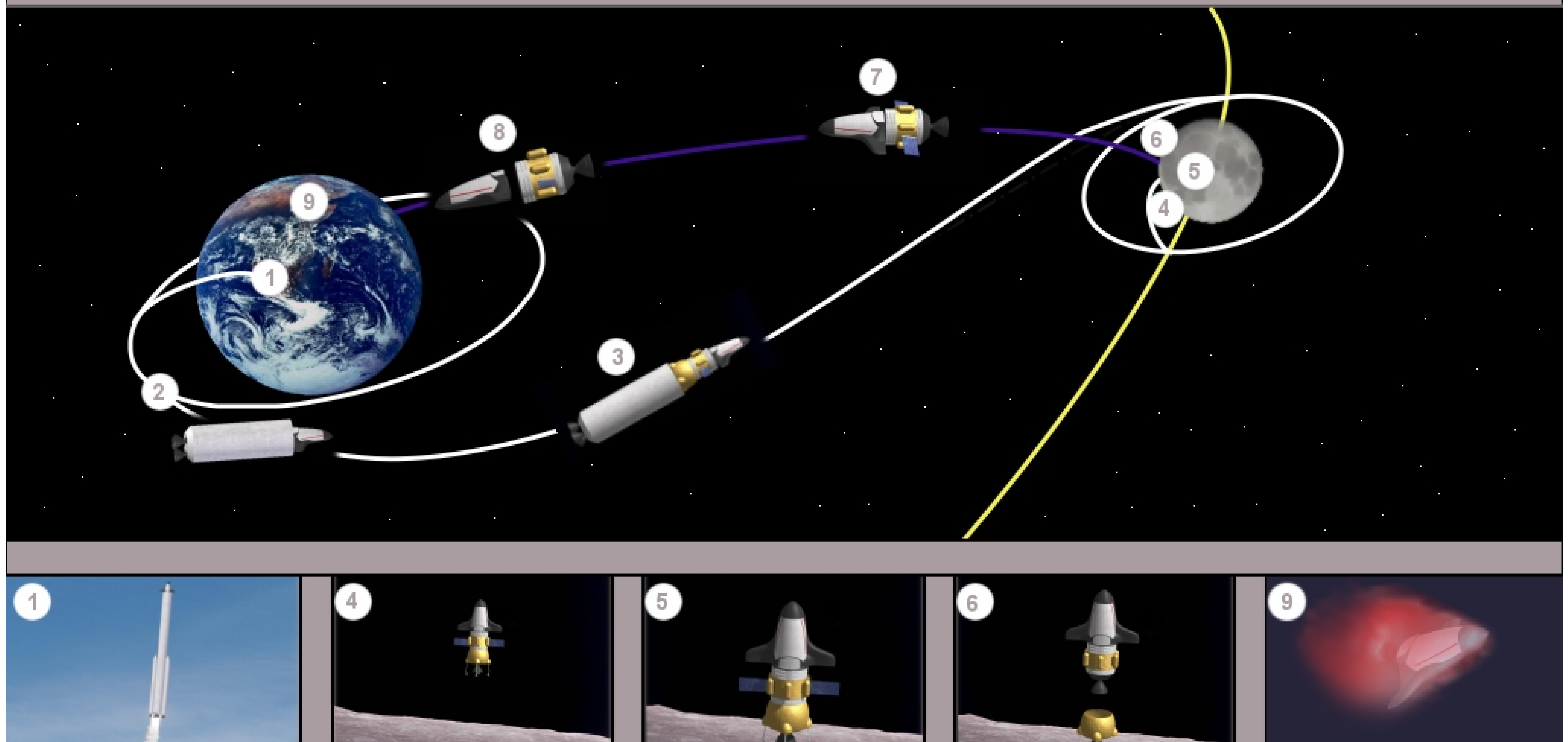
Phase 5: The whole system lands together, with the Command Module (CM) at the highest level, the Service Module (SM) at the middle level and the Lunar Module (LM) at the lowest level. The LM supports the other two levels.

Phase 6: After seven days the CM and SM launch. The LM is left on the surface and used as a launching pad.

Phase 7: The two systems directly enter into the Earth Return Orbit (ERO).

Phase 8: As the spacecraft approaches Earth it jettisons the SM and the CM.

Artemis Flight Sequence



1. Launch of spacecraft and EDS stages from Kennedy Space Center
2. LEO insertion of launch vehicle
3. Completion of TLI burn
4. LLO phase
5. Landing on the Moon
6. Liftoff from lunar surface
7. TEI insertion
8. Seperation of CM from SM
9. Reentry into Earth's atmosphere

Phase 9: Atmospheric entry of the CM and makes its landing at Edwards AFB, CA.

This architecture enhances safety through three main features. The first and most obvious feature is the direct return element from the moon to the Earth. A direct return means that the spacecraft launches from the moon and goes back to the earth without having to do any Lunar Orbit Rendezvous (LOR). This increases safety in case of an emergency on the moon, because all critical return modules are already in place. This also means there is a quick and reliable way to get back to Earth. Another feature of the architecture is the lack of any LOR's during the mission. Most other lunar architectures, including Apollo, have some sort of LOR. Team LETO decided that having a safer spaceflight was most important and avoiding LOR was the safest approach. Also, reducing the number of rendezvous inherently increases reliability, and the lack of any docking maneuvers in lunar orbit further escalates the safety factor. The last feature mitigates an issue that plagues most other EOR-Direct missions. The problem lies in the inability to have any sort of "Apollo 13 option". This refers to the ability to have a lifeboat other than the CM. Team LETO mitigates this problem by equipping the Service Module (SM) with habitation capabilities that will be used more often during the lunar operational phase of the mission.

Other architectures that were considered were Apollo style architectures, EIRA and Tempest. The pros and cons of these different kinds of architectures can be seen in Table 1. The first column shows that the EOR-Direct architecture has the same number of launches as EIRA, This means the reliability of Artemis launches is on par with NASA architectures. The second column shows that the number of earth rendezvous is less than any other system. The final column shows that no rendezvous are done at lunar orbit. As compared to other architectures the EOR-Direct has the highest advantage in terms of safety, as well as equal or greater reliability.

3.2.2 Baseline Vehicle Description

Project Artemis' spacecraft was chosen based on the mission architecture, safety and reusability.

Team LETO decided from trade studies done on mission architecture that EOR-Direct would be the best option. Keeping

Architecture	Launches	Rendezvous	LOR	Lifeboat
Apollo	1	2	1	Yes
ESAS	2	2	1	Yes
EIRA	2	2	2	Yes
EOR-Direct	3	2	0	Yes
Tempest	3	3	2	No

Table 1: Artemis trade study highlighting the reasons for choosing EOR-Direct.

this in mind, as well as safety, the spacecraft needed to have the capability to fit into the architecture as well as provide a safe vehicle for the crew. It was decided that a mesh between an Apollo style and Soyuz style vehicle would best suit the mission. The vehicle can be seen in Figure 1.

This vehicle consists of three different modules. They are the CM, SM and LM. The three module design is similar to the Soyuz vehicle. The main difference between Project Artemis' spacecraft and Soyuz is that the SM and Orbital Module in Soyuz are combined into one module in Project Artemis. The third module was inspired from the Apollo architecture. The LM simulates the descent stage of the LSAM from Apollo. Some of the differences include an extra need for structures due to the load weight of the upper components and extra storage capability for payload. The equivalent of the ascent stage of the LSAM would be the CM and SM.

The reason this system is safe is due to the capability to return to Earth even if there is a problem with the LM. Also, because there are two living spaces the crew can stay in either module. In case of an emergency, albeit, the crew can only return in the CM. The main TPS system of the spacecraft is not connected to any other modules. This makes the CM safer in case of any explosions from other modules, such as in Apollo 13.

Some of the trade-studies conducted included a single vehicle design (SVD) that would be able to accomplish all the goals that the Artemis architecture requires. The Apollo Design (AD) approach was also considered, using the upper stage of the LSAM

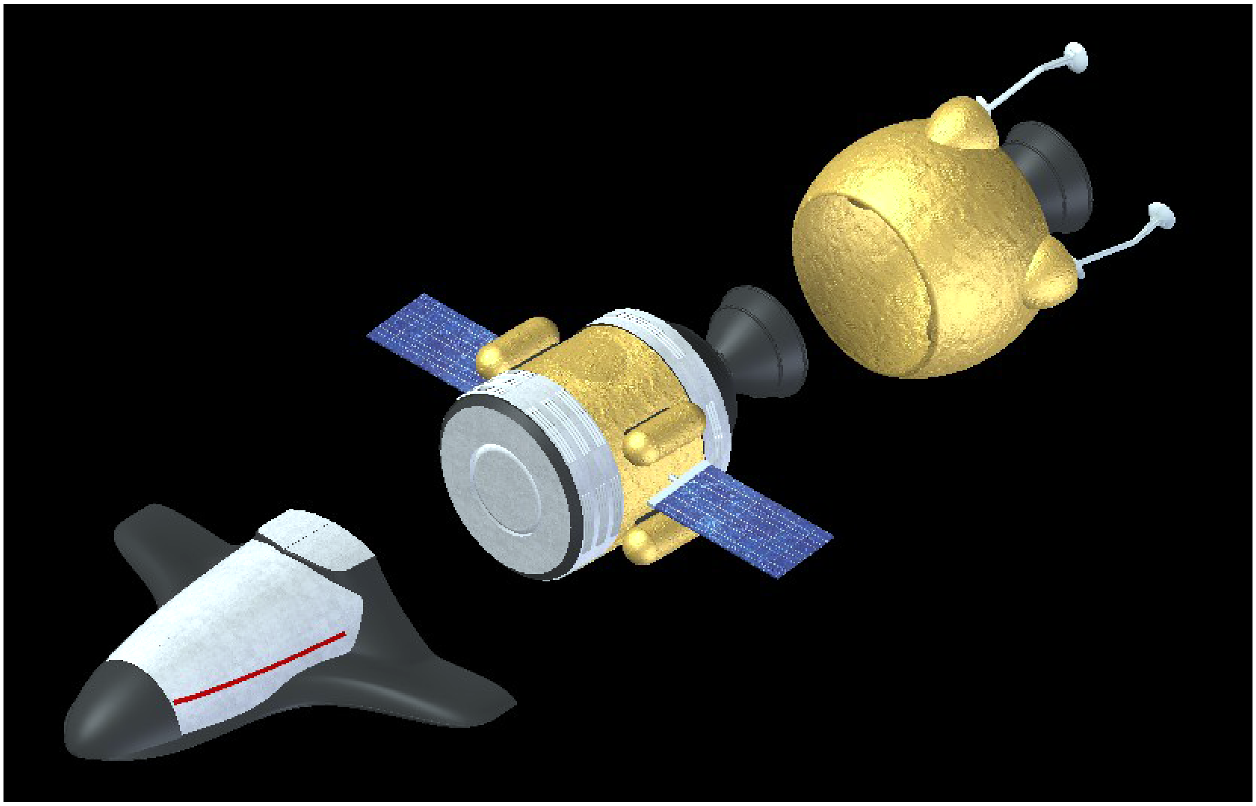


Figure 1: Artemis spacecraft in a blown out view highlighting the different modules from left to right: CM, SM, LM.

to directly insert back to Earth. A Reusable Landing Module (RLM) was also considered, in which a module is designed that can be reused for future landings. The last kind of overall vehicle designed was the Artemis design.

The RLM was deemed unsafe as well as unreliable and would also be quite massive. The unreliability comes about with the multiple usage of a vehicle that would stay in orbit for a period of 10 years. Extra docking procedures would also have to take place, be they EOR or LOR. Also, to fit into the architecture the lander would have to ascend with the spacecraft and come back to Earth orbit, which would greatly increase mass. The SVD approach is the most obvious for an EOR-Direct mission, but was found to be very massive and would also not have Apollo 13 capability. The AD method was also considered but would also fall into the same problems as the other concepts; It would be very massive because it would require two large life support systems and dual subsystems to launch back to Earth. The Artemis design was chosen because of the lower possible mass and the higher reliability.

3.3 Launch Sequence

An EDS is launched up into LEO. It loiters for up to 45 days awaiting the LTS and second EDS. The CaLV deposits the LTS/EDS into 56x 296 km at 28.5-deg where they can dock with the awaiting EDS.

Project Artemis requires two launches in order to get the LTS and two Earth Departure Stages (EDSs) up into LEO. Team Artemis' chosen LV, the CaLV, is launched at pad 39 at the Kennedy Space Center (KSC). Team LETO used ESAS to approximate the launch schedule of this LV when launching the LTS and two EDSs. First the EDS used for TLI is launched into orbit awaiting the LTS and the final EDS. It sits in orbit for up to 45 days before it finally can attach to the LTS and remaining EDS. It follows roughly the same launch sequence as the LTS/EDS launch. Figure 2 shows both launches.

The two RSRBs are ignited at launch with the five RS-25s of the core engine. After burning for 132.5 s, the RSRBs separate from the core vehicle at around 47 km and coast to an apogee of about 73 km. They launch parachutes and land in the Atlantic Ocean, where they are recovered for reuse. After the RSRBs separate, the core stage burns for an additional 275.5 s, causing the payload to reach Mach 12.12 and is jettisoned at an altitude of 124 km. The core stage then enters a suborbital ballistic orbit, enters the atmosphere, and its debris lands in the South Pacific Ocean. About 39 s after the core stage jettisons, the shroud and LES jettison from the LTS (just the shroud when just an EDS is launched). Once the core stage separates the EDS burns for 218 s to provide the final impulse into LEO. The total delta-v at this point is 9260 $\mathrm{m \cdot s^{-1}}$. The LTS and LOI EDS is now inserted into 56x 296 km at 28.5-deg where it docks with the awaiting TLI EDS.

3.4 Concept Originality

Team LETO understands and fulfills the RFP's desire for original concepts while keeping a realistic and practical solution to return to the moon.

The relative importance of originality, reusability, safety and cost was assessed using AHP. After coming to a cosensus that

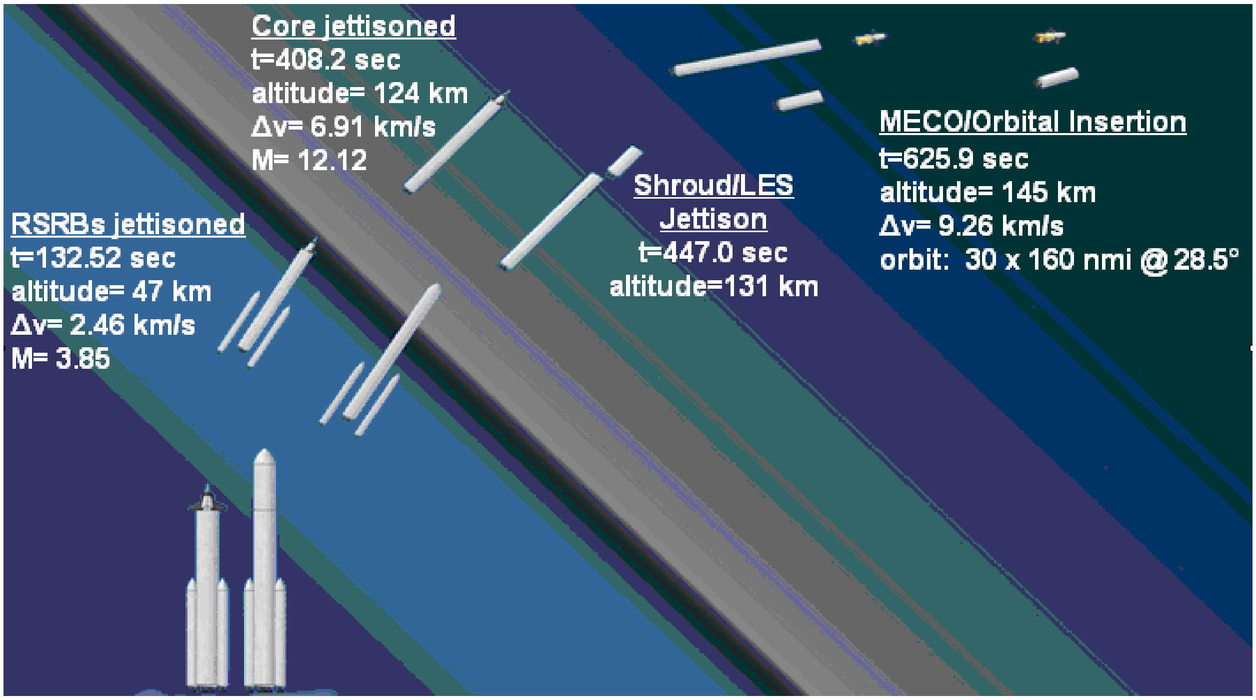


Figure 2: Launch Sequence for LTS/EDS and EDS.

safety was the most important theme, Team LETO proposed several different views for a unique solution to the RFP. One of the original concepts proposed was using a single spacecraft that could perform all required functions of the architecture. The lander, command module and service module would all land on the moon as one package. The reason for this concept was to reduce architecture complexity issues such as rendezvous and increase safety.

Another concept that was proposed was to make the Artemis vehicle as modular as possible. By keeping the vehicle modular the ability to refurbish and have a quick turn around time would increase. Even though the system will land on the moon together they will still be separate. The reason for this is to shed mass during lunar launch and earth reentry.

A final factor of uniqueness in the design comes from the use of a moderate winged body design for the body of the entry craft. The positive aspects of this design are that the CM can be reusable and have a controlled reentry flight.

Team LETO's architecture was formed around the safety issue, so possibilities such as nuclear power and ceramic tiles were not included into the project, but can be seen in the different trade studies the team analyzed. To maintain high level of concept originality various new technologies for subsystems were chosen, but are for the most part heritage technology.

3.5 Technologies Employed

Artemis uses a variety of different technologies, many of them being in the forefront of their fields. These technologies will all be proved before manned missions.

Team LETO looked at various manned missions, from Apollo to the Space Shuttle. By inspecting the subsystems employed in these systems Team LETO gained an initial foothold for what to look for in terms of technologies, as well as seeing what needed improvement and what subsystems could be changed completely. The TRL levels for the various subsystems and components can be seen in Table 2. The majority of the subsystems have high heritage technology such as the GNC, power and TCS subsystems.

Technology	TRL
Aerojet 445	9
Airlock	6
ARMOR	6
Comm. Systems	9
GNC System	9
LES	7
Life Support Systems	9
Li-ion	9
Other TPS	9
Parafoil	7
Proton 100K	8
Radiation	8
RS-72	7
Space Suits	6
TCS System	9
UTJ GaAs	9
Vacuum Cleaners	7

Table 2: Technology Readiness Levels (TRLs) of various Artemis components.

TRL levels were chosen based on the NASA TRL classification. Any system that has been flight proven is at a TRL of 9. Technologies that have been checked off by NASA are at level 8. Highly developed technologies are at level 7. All other systems are at 6. These systems have been tested before and need improvement to achieve a higher TRL level and thus increase reliability and safety to a satisfactory level.

The propulsion systems used are based on existing systems that are already designed and built by various companies. There are a variety of improvements that were done in order to scale the main engines for the mass constraints encountered with Artemis. The EVA subsystem uses vacuums for dust mitigation. This system has never been used on the moon before. The GN&C components are flight proven technology that did not have to be improved from existing systems. The TCS system consists of various different kinds of subsystems, all of which have been used before on either non-manned or manned missions. The TPS system technologies already exist and have been used, sans ARMOR. ARMOR is at a TRL level of 6. The power subsystem components are all flight proven technology as well. The life support systems are all basic and have been used on all other manned missions. The radiation subsystem has a component that has never been used before. Although the TRL level is low, the value should increase quickly since it is a passive material component. Also, the structures subsystem does not feature new or unproven technology that will be used either in terms of materials or in structural type.

The choices for various subsystems was enforced due to safety concerns, not just performance. The life support systems were chosen because of their high TRL levels as well as the short mission time. ARMOR was chosen because it has the capability to far exceed the safety of the troublesome shuttle tiles. Technology for the vacuum cleaners is also important due to their need to lessen the damage to spacesuits and abrasion in the interior.

4 Technical Approach

4.1 Design Methodology

Analytic Hierarchy Process (AHP) and the Design Structured Matrix were extensively used during the course of the project to determine & optimize the individual subsystems for the Project Artemis.

A standard system engineering process was employed by team LETO at various stages of the design to select the best possible design solution.

The morphological matrix, as seen in Figure 3, was developed during the preliminary stage of the design process, it included the different solutions possible for each individual subsystems that were studied in detailed during the course of the project.

Team LETO used AHP to arrive at the

Subsystem	Alternatives					
	1	2	3	4	5	6
Mission Architecture	Apollo	ESAS	EIRA	EOR-LOR	EOR-Direct	Tempest
Launch Vehicle	4 segment RSRB with 1 SSME	Atlas V	Delta IV	Space Shuttle Derivatives	5 Segment RSRB in line with 5 SSME Core-Cargo.	
Propulsion	Solid	Liquid	Solar	Nuclear	Ion	
Power	RTG	Nuclear Reactor	Solar Pannels	Fuel Cells	Chemical Power	Lithium Ion Batteries (ESAS)
Communication	S-Band	Ku-Band	Ka-Band	Optical System		
Life Support System	ESAS	ISS				
Lunar Descent	Single Stage	Apollo	Two Stage Jettison			
Earth Landing	Parafoil	Parachute	Winged body freefall			
TPS	Carbon-Carbon	A.R.M.O.R	AFRSI blankets	HRSI Ceramic tiles	Aerogel	
TCS	Passive	Active				
GNC	IMU	Optical Navigation	Lunar Orbiters			

Figure 3: Team LETO was able to study different possible solutions from different alternatives presented in the morphological matrix to select an optimum solution.

best possible solution for most of the trade studies.

A baseline design was selected using morphological matrix, which was then compared to the different possible solutions using the Analytical Hierarchy Process to obtain an efficient and optimized solution.

Another widely used technique in space system engineering community, DSM, was use to capture the interdependencies between different sub- system elements. This tool was very beneficial while building the optimization tool.

4.2 Automated Design Tools

Project Artemis Recognized the importance of creating tools to determine mass, power, and other estimates for many subsystems. Here is a list tools used for this project.

Communications Comparison Tool A complex design tool that constructs the communications subsystem for different bands, bandwidths, and antenna sizes under specified power constraints.

Lunar Breakdown Tool The Lunar Work Breakdown tool was created to be used for

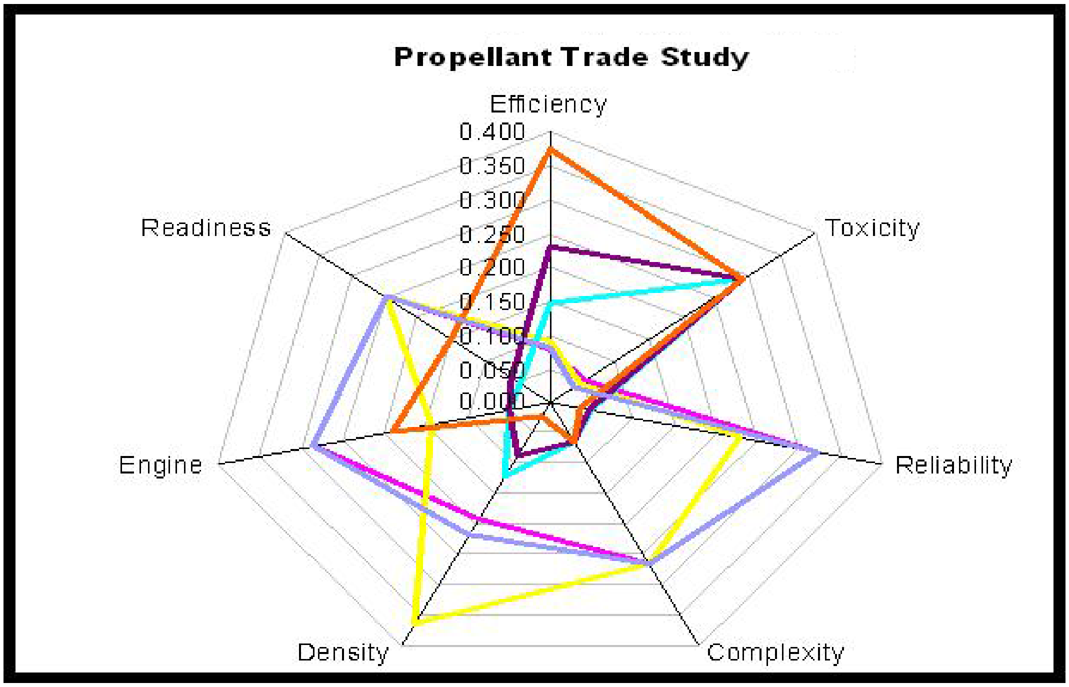


Figure 4: Propellant Trade Study using AHP shows that for this study Reliability and Readiness level were one of most important criteria.

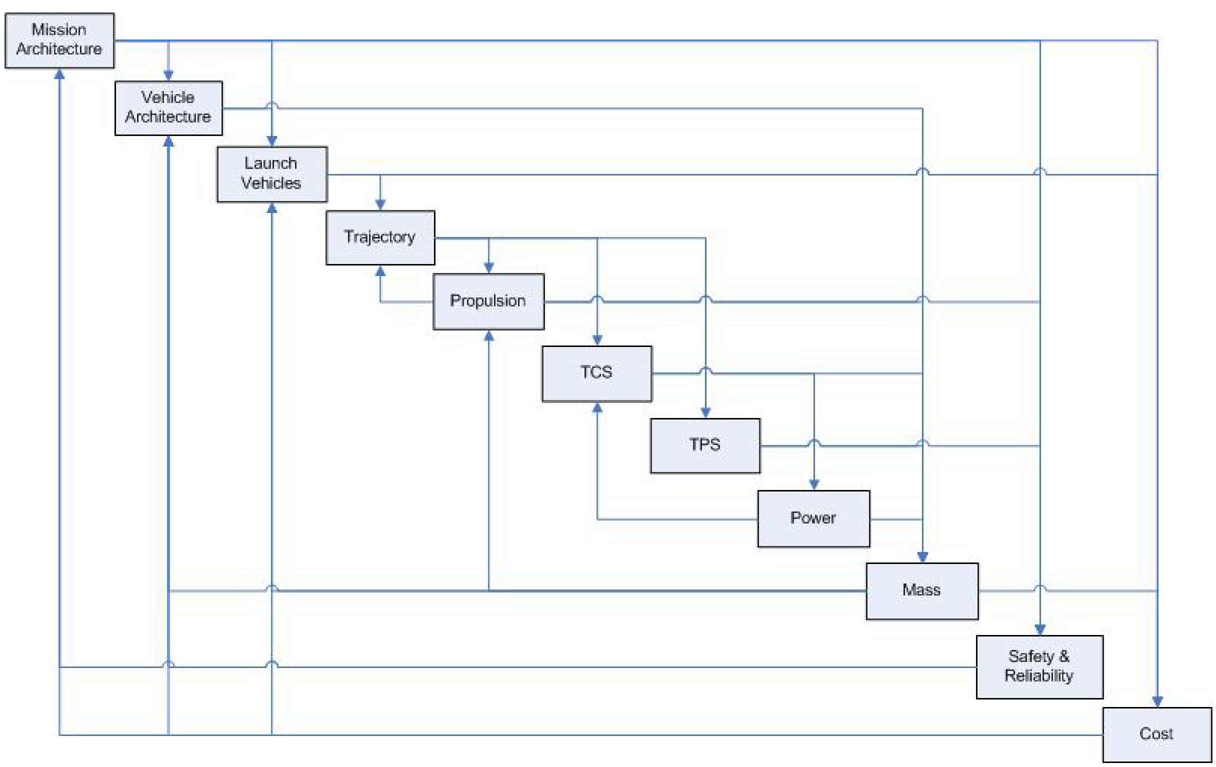


Figure 5: DSM was used to identify interedepencies between different subsystem, which later helped Team LETO to optimize the vehicle.

a system wide calculation of the mass and power of the entire Artemis project. This includes the dry and wet masses.

TCS Tool The TCS tool calculated the heat in and heat out during the phases of the mission. It calculated the heat dissipated by the passive system and the required active heat dissipation further needed. This was used to calculate the size and mass of the radiators and the heaters. Then the total mass and power requirements for the TCS was calculated.

TPS Tool The TPS tool calculated the mass breakdown of all materials used for the TPS and the total mass of the TPS system.

Propulsion Mass Calculator (PMC) The Propulsion Mass Calculator is used to calculate the amount of propellant needed for the orbital maneuvering system (OMS), the reaction control system (RCS), and the propulsion subsystem mass. It was also used for the tradeoff studies of whether to use two EDS stages or one EDS stage for propelling the LTS to LLO. The results showed that using one EDS stage will be too massive for any current launch vehicle.

Spacecraft/Vehicle Level Cost Model This tool was used to determine production, development, and total cost of developing the launch vehicle. The program uses dry mass, number of launches, and a learning curve to determine these costs. The SVLCM is a top-level model derived from the NASA/Air Force Cost Model (NAFCOM) database.

5 System Elements

5.1 Command Module

The command module is a lifting body that is completely reusable. To enhance safety a known aerodynamic body is used with new and better thermal protection.

The Artemis Command Module (CM) is a lifting body in the heritage of the Space Shuttle and X-plane lifting bodies. Lifting bodies have been studied for many decades and are a well-known design for supersonic flight. Team LETO decided that a lifting body would also be best because it is a safe method for return, as well as a highly reusable vehicle. A cutaway view of the CM can be seen in Fig-

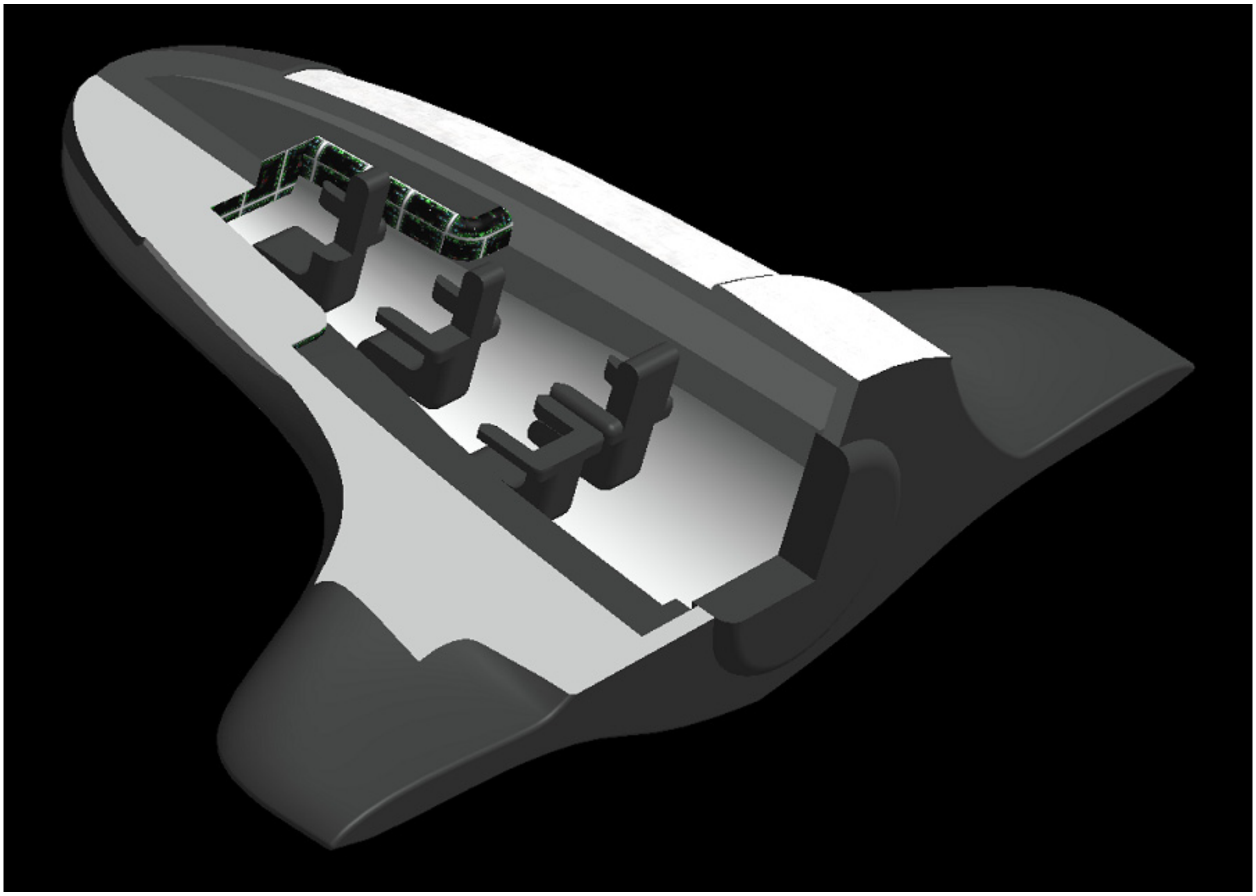


Figure 6: Cutaway view showing the winged body concept.

ure 6.

The subsystems on the CM include all life support, attitude control, communications, command and data handling, guidance and navigation, reentry and some power and thermal systems as well. All the subsystems that return have a positive impact on the reusability of the system as a whole. The CM is also the main living space for the astronauts for launch, transfer and reentry. Team LETO has also met the requirements in the RFP by including enough space for four astronauts as well as payload. The non-human payload for the system is designed to carry 500 kg, but only 100 kg will be carried in the CM.

The subsystem that takes up the majority of the space in the CM is life support. The reason for this being the large mass (2546 kg). Also, life support is a system that must be used throughout the whole mission so it needs to be located in the CM. Some of the life support functions are also placed throughout the SM as well. The subsystem is placed in the rear of the CM to make more room for human activities and to free up space for computing components. The crew will be placed roughly in the center of the CM. The payload space is located in the rear. GNC, C&DH and its backups are all placed towards the front of the CM. The parts of the power system that reside in the CM are the PMAD system and the Li-ion batteries. These systems will be placed towards the front. The CM AC&D consists of 12 thrusters, 6 on the wingtips and 6 on the nose. The communications are placed on the rear of the CM. The thermal system is placed all around the outer part of the CM. The CM needs to be protected at all areas from the environment. The reentry system also has to be included with the CM. The parafoil will be placed on the top near the center of mass and the majority of the TPS is placed on the bottom of the CM.

The CM increases the reusability through high modularity. The modular parts include life support and all other subsystems of the CM. The increased modularity of the system brings about an ease of testing, installation and removal. All these aspects cut the cost in turn around time. The safety comes about with accessibility of parts by the maintenance crew and their ability to see problems with more ease.

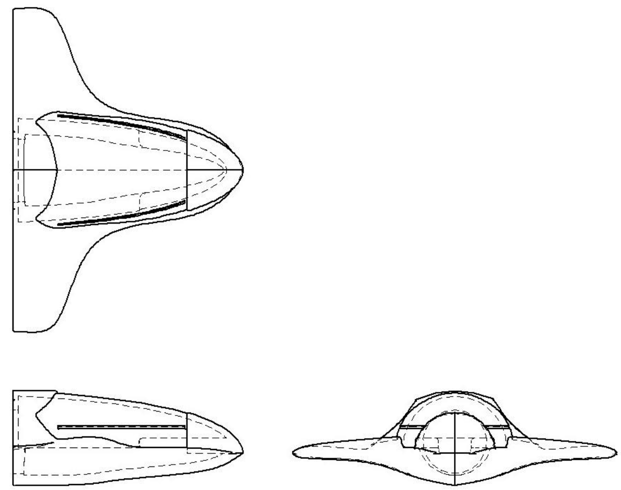


Figure 7: Multiple view of CM highlighting the winged body heritage design making the CM reliable.

The winged body shape seen in Figure 7 shows that the CM takes its heritage from such designs as the Shuttle and the Russian-designed Kliper.

5.2 Thermal Control System

The Thermal Control System will be composed of an active and passive system, involving MLI blankets for the Service and Lunar Module, TPS material for the Command Module, radiators, Coolant loops and heaters.

The Thermal Control System (TCS) has

an important role of maintaining a delicate balance of the temperature inside the spacecraft. The temperature during the mission ranges from a freezing -160 C of space to extremely high 1650 C during re-entry. Different subsystems of the spacecraft have a different temperature range in which it must be kept for it to function. Also, due to it being a manned mission, an ideal temperature of 297 K is recommended. A small range of temperature must be maintained to ensure that the astronauts are comfortable and safe for the duration of the mission.

TCS can be managed in two ways; passive and active control. Passive control is the means of controlling the temperature with no moving parts, or power. This usually involves the use of paints and radiators to reject heat and insulators to keep the heat in. This is a highly effective system and is used for most cases. Active control is needed when a very precise means of control is needed, such as a human mission. Active control is an addition to the passive control in which heat is moved around and ejected through moving parts and power requirements. This uses systems such as heaters, coolant loops, water evaporators, and heat pipes.

For the spacecraft, a combination of active and passive system are used because it is a manned mission. The command module's passive TCS is its TPS protection doubled as the TCS system. The TPS system is covered in a later section. It provides the necessary insulation needed during the duration of the mission. Multi-Layer Insulation (MLI) Blankets are used to cover the service module and the lunar module for insulation, and radiators are placed around the service module to dissipate excess heat from the spacecraft. The radiator is sized to $44.7\,\mathrm{m}^2$ to dissipate 17.9 kW of energy during the hottest part of the mission, which takes place during near earth orbit. During this time, the spacecraft takes in 56.8 kW of energy from the sun and 17 kW of energy from the Earth's albedo, while heat dissipated from the passive system of the spacecraft is 63.9 kW. The remaining energy is dissipated by the radiators.

The active system involves an internal TCS and an external TCS. This helps regulate the temperature all around the spacecraft to its desired setting. The internal TCS is a water coolant loop which connects to

Equipment	Operational Temp	Survival Temp
Communication	-30–55	-10–45
Battery	0–25	-10–25
Tanks	-10–50	-10–40
C & DH	-20–60	-40–70
Solar Panels	-150–110	-200–130
Units	C	C

Table 3: Shows the temperature range of various components of the spacecraft.

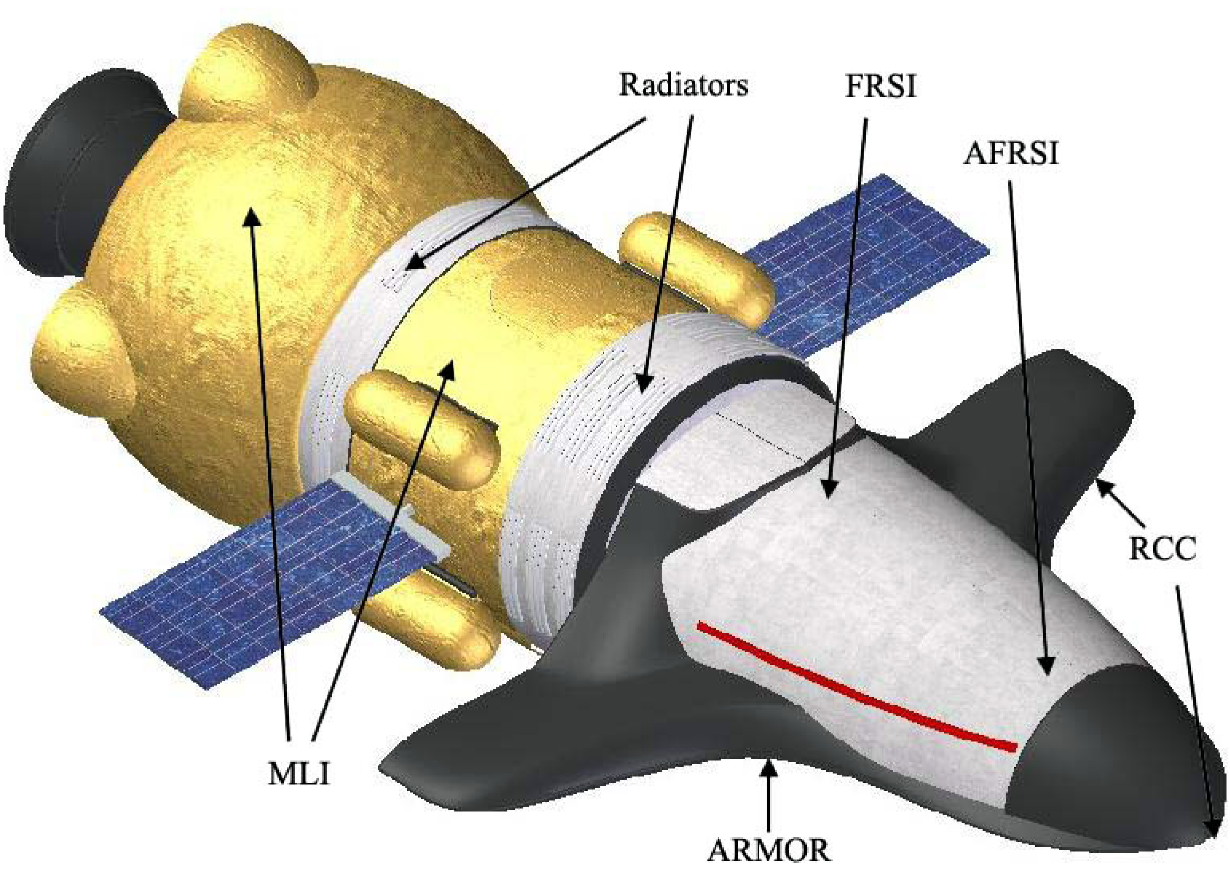


Figure 8: Shows the passive TCS systems of the spacecraft.

various components of the spacecraft. The coolant loop will take away heat from the components and sends it to heat exchangers located in various locations throughout the spacecraft. The heat exchanger exchanges the heat from the internal coolant loop to the external TCS. The external TCS involves an ammonia coolant loop which takes the heat from the internal coolant loop via the heat exchanger and relays it to the radiators to disperse the heat into space. The coolant loops are connected to the spacecraft's command and data handling, so that the astronauts are able to control the rate in which the heat is removed from the systems.

The internal TCS also includes heaters to heat the spacecraft during the coldest stage of the mission. During the lunar trajectory stage, 6.3 kW of heat needs to be generated to keep the ideal temperature for human life. This heat is generated by electrical heaters on the spacecraft, which are fully controllable by the astronauts. This system takes up to 144.8 W of power and have a mass of 44 kg.

The TCS system works throughout the mission to maintain the ideal temperature needed for human life. With the astronauts able to control the coolant loop and the heaters, there are multiple ways to maintain the temperatures the astronauts feel the most comfortable in. This provides a more reliable and safe method of TCS to aid the astronauts throughout their mission.

5.3 Life Support

By incorporating redundant and innovative life systems, Project Artemis ensures crew survivability throughout the duration of the mission.

A major requirement of any vehicle that is to travel through the vacuum of space is that the system be able to protect its passengers from the harsh environment that it is traveling through. For this reason, life support is one of the most critical systems on any manned space flight.

Life support is broken down into two several categories. First, it must provide for the crew, and second it must remove potentially hazardous substances from the system.

To provide for its astronauts, Project Artemis stores the food it will need for the entire mission in an open cycle fashion. This decision was made due to the relatively short duration of the mission and the low mass of food required for this type of mission. Based on historical data, the average astronaut consumes 2.3 kg of food per day. Based on a fifteen day (longest possible duration) mission, Project Artemis carries only 138 kg of food for the entire mission. To complement the food requirements, a single microwave oven and sink are built into the LTV. This allows the astronauts to move away from the “liquid meals” that have been in use by the space program for a number of years. To promote hygiene a shower is also built into the LTV. The shower does not use running water, but rather allows the astronauts to use a “sponge bath” technique.

Figure 9: Project Artemis’ toilet system is similar to the one that has been in use on the space shuttle for over two decades.

The toilet also uses very little water in its operation. Waste does not flow in the absence of gravity, therefore air flow techniques must be used to remove waste away from the body. The solid waste that is collected from the WCS is stored onboard until reentry. Liquid waste is periodically vented into space.

The sleeping provisions that are given to each astronaut consist of cot-style bedding that fold out of the way when not in use. Each of these cots has a mass of about 9 kg.

Water is stored in tanks attached to the outside of the service module. The total water requirement is approximately 1400 kg. This amounts to about 24 kg of water per person per day.

Air is also stored in tanks on the outside of the service module. Air, unlike waste and water, is operated on a semi-closed cycle. This means that some of the air is recycled and reused.

Expelling potentially harmful substances from the cabin atmosphere is another vital

Figure 10: LiOH canisters used by Project Artemis to remove the poisonous CO_2 gas from the cabin atmosphere.

Atmosphere	Requirement
Total Pressure	99.4 kPa
Partial Pressure O_2	20 kPa
Partial Pressure CO_2	0.4 kPa
Partial Pressure N_2	79 kPa
Temperature	297.15 K
Humidity	50%
Ventilation Speed	$0.15\,\mathrm{m\cdot s^{-1}}$

Table 4: Project Artemis uses the above values for its atmospheric control and monitoring systems.

task of the life support system. Lithium hydroxide filters are employed on Project Artemis to remove the poisonous carbon dioxide gas from the cabin atmosphere. These filters are placed in strategic locations throughout the cabin. They also need to be changed on a regular basis. This is done as part of a daily routine by the crew. A small fan is included as part of this system to promote airflow through the system. After the carbon dioxide has been absorbed by the canister, the clean air is cycled back through the system.

Project Artemis employs the use of sensors and mass spectrometers for atmospheric monitoring. This active system is able to alert the crew if something is not right. This is done through the use of caution and warning lights that appear on the control panel.

As shown in Table 4, Project Artemis uses a relatively low atmospheric pressure (Earth's sea level pressure is approximately 101.32 kPa). There are many advantages to this. The most important one is that a lower pressure eases the transition into the pressures of the EVA suits (27 kPa). If the pressure inside the cabin were higher, a longer pre-breathing exercise would be required before exiting the airlock, thus wasting valuable time. Lower pressure also makes the cabin less susceptible to fire, a major concern in space.

The partial pressures of different gasses used onboard the LTV are identical to the ones occurring naturally on Earth. Oxygen and nitrogen are stored onboard, and carbon dioxide, which is expelled during breathing, is controlled to remain at 0.4 kPa. Control of trace gasses occurs similarly, but through mass spectrometers described above. Finally, air is circulated around the cabin at relatively low speeds. This is to prevent the atmosphere from becoming "stale", and allows the astronauts to have "fresher" air, and to make sure that air flows through the LiOH canisters.

Project Artemis's life support system is a highly innovative system that is responsible for protecting the crew throughout their fifteen day mission to the lunar surface.

5.4 Lunar Extra-Vehicular Equipment & Support

By incorporating redundancies into EVA systems of Project Artemis, crew safety is enhanced dramatically during one of the most mission critical phases of the operation.

The RFP requires that the LTV be able to transport 400 kg of cargo to the lunar surface and 100 kg of cargo to return to the earth. Secondly, the crew must be able to perform EVA's on each of the seven days that the astronauts could conceivably be on the moon.

The Apollo program was forced to deal with the previously unforeseen effects of lunar dust on the EVA suits and the LSAM itself. The dust caused premature abrasion to the suits and instruments inside the LSAM as it was tracked in from outside. This problem was unforeseen during the Apollo program, however, Project Artemis will combat this problem.

Given the requirements of the RFP and trade studies based on these requirements, the LTV does not have a lunar roving vehicle (LRV) built into the ship's design. However, it is flexible enough to allow for an LRV to be transported to the lunar surface should the mission require it. This vehicle has to be designed to be less than 500 kg so that it can brought along in the cargo bay.

After doing trade studies based on the length of time the astronauts will spend on the lunar surface, the number of astronauts involved in the mission, and the availability of present day technology, it was determined

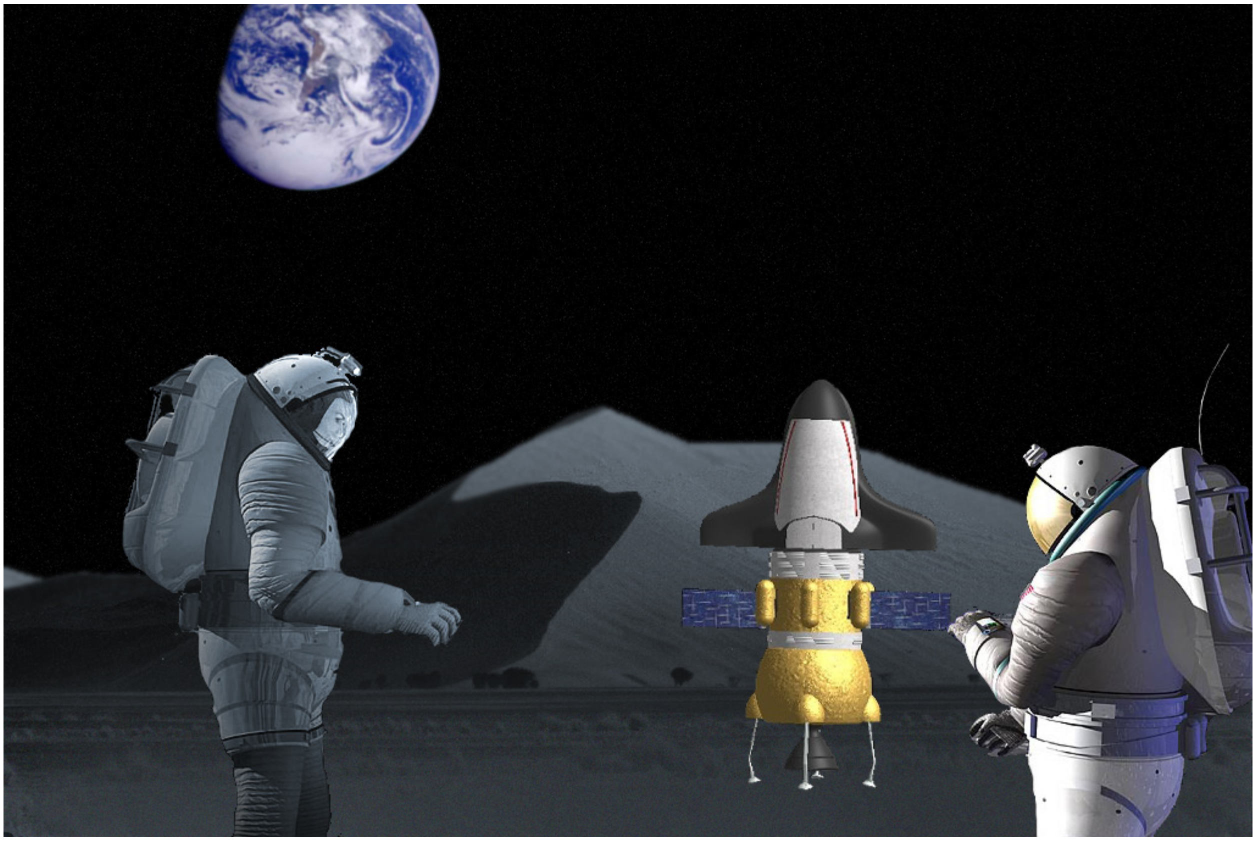


Figure 11: Astronauts exploring the lunar surface with the LTV in the background.

that it would be unreasonable for the crew to depressurize the entire cabin each time an EVA were performed. To facilitate this requirement, an airlock has been designed as an integral part of the LTV. This airlock allows for two astronauts to exit the LTV at any particular time without having to depressurize the cabin. This makes the mission itself safer in that the interior cabin space is not subjected to constant depressurization and repressurization. The greatest advantage is that it allows the astronauts to work in shifts (if the mission allows), thereby increasing the amount of time actually spent on the lunar surface.

Given the fatigue factor of working in a heavy spacesuit, the maximum amount of time an astronaut can be expected to perform a lunar EVA is about eight or nine hours. If two teams of two astronauts are used, the time spent on the surface can be up to eighteen hours per day. While one crew is on the lunar surface, the other crew can sleep and monitor the ship. This is a great advantage

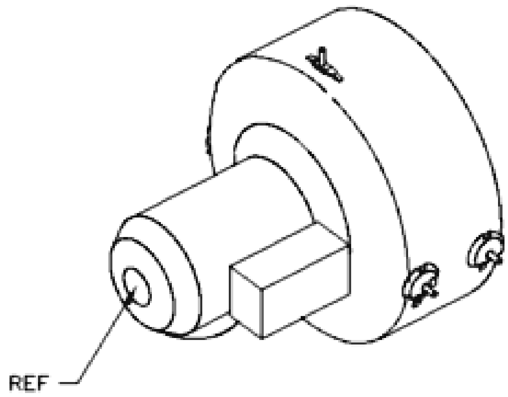


Figure 12: Schematic drawing of Project Artemis' airlock design.

compared to all four crew members being required to exit the vehicle at the same time.

The airlock is mainly composed of aluminum and has a mass of 275 kg. It is designed to completely depressurize in seven minutes. This does not include a required thirty minute "pre-breathing" exercise performed by astronauts prior to EVA activities. To begin, two crew members enter the airlock and begin their pre-breathing routines. While they are performing this task, they are simultaneously donning their EVA suits, which are stored in the airlock. After the pre-breathe exercise is complete, the airlock is depressurized and the astronauts are free to exit the vehicle. The airlock can be operated from both inside the airlock and from the cabin.

The EVA suits are the final component of the EVA system. The suits are based off the suits currently in use by the International Space Station (ISS). It operates at a pressure of 26.35 kPa and has a mass of 30 kg. This does not include the astronaut's portable life support system which has an additional mass of 15 kg. These suits provide a much greater range of motion than did either the Apollo suits or the Space Shuttle's suits. Project Artemis stores the five suits in the airlock. The fifth suit is for redundancy in case something goes wrong with one of the others.

The suits have many necessary provisions already built into the suit's infrastructure. A visor is built into the helmet to reduce the glare that the sun can cause. This is analogous to wearing sunglasses to reduce glare while driving an automobile. Lights are also built into the helmet to allow the astronaut to have some visibility while working in a partially shadowed area. Another provision is a camera built into the helmet. This allows the astronauts to review their surface missions long after they have returned to the LTV. Finally, the suit is temperature controlled and contains a drink bag so that astronauts have a source of water during their EVA experiment. These are just some of the many provisions provided by the space suits designed for Project Artemis.

In an effort to combat the lunar dust problems encountered by the Apollo program, Project Artemis employs the use of vacuums in the airlock. After returning from the lunar surface, the astronauts will turn the instruments on and attempt to remove some of the regolith from their suits. Finally, the since the EVA suits are stored in the airlock, there is no need for them to ever enter the cabin. This prevents dust from ever entering the cabin.

The EVA suits are the only equipment needed for EVA that is actually built into the LTV. Other equipment such as a LRV, shovels, or other miscellaneous tools must be dictated as necessary by the mission planner and included as part of the 500 kg of cargo that Project Artemis is capable of transporting to the lunar surface.

The EVA system is clearly distinguished by its adherence to the “safety first” policy that Team LETO has adopted. These systems are the safest in the industry due to the airlock’s design, EVA suit design, and the redundancies of each component of the system.

EVA	Mass	Power
Airlock	275	100
Suits	225	0
Dust Mitigation	10	100
Total	510	200
Units	kg	W

Table 5: Mass and power breakdown of the EVA system.

5.5 Payload Storage & Delivery

The 500 kg payload to the moon is stored in a self-contained corner of the LM on a movable platform while the return cargo of 100 kg is stored in the command module.

The scientific instruments are stored in especially designed containers that are able to sustain a force of 1500 N. Other less sensitive cargo is stored in normal rectangular metallic containers capable of sustaining a force of 1000 N. This design is for emergency scenarios when jettisoning the cargo is necessary.

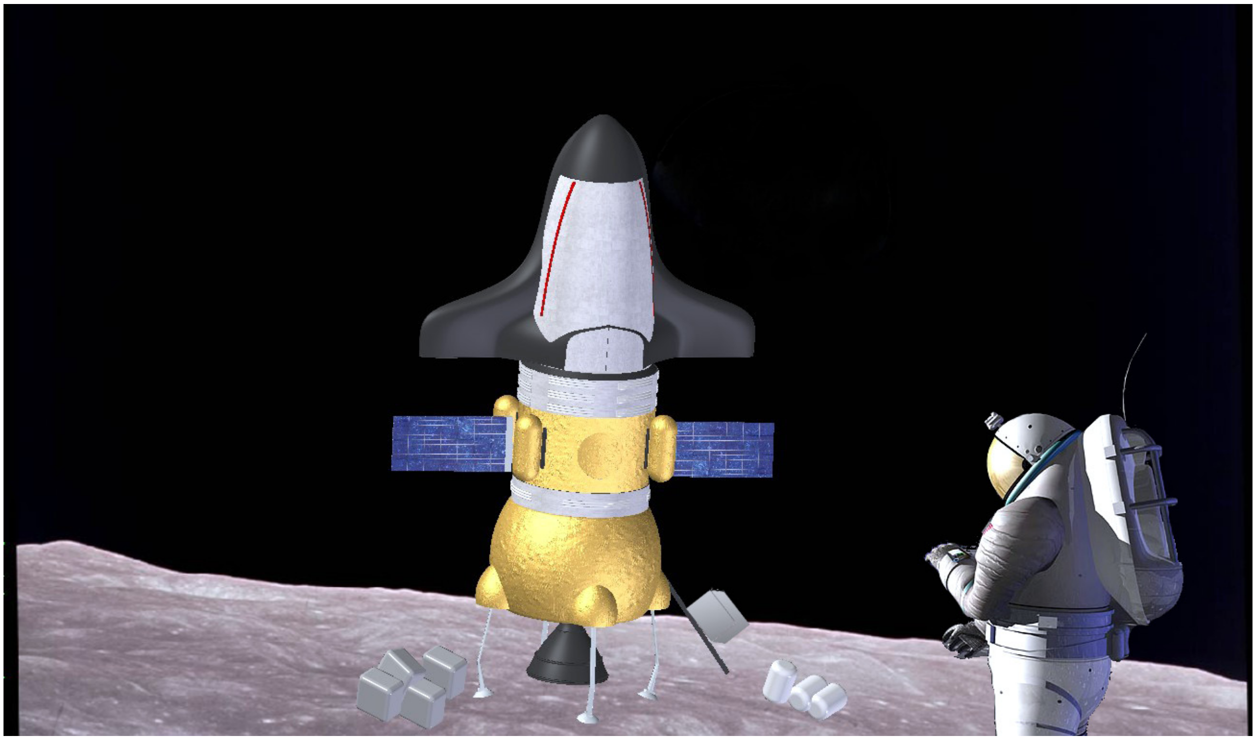


Figure 13: Unloading cargo on the lunar surface.

Upon arrival on the moon, the platform moves the cargo to the exterior of the module by first going through an airlock. This is in case of pressurized material.

For the payload delivery, the astronauts have shovels, picks, and other assorted tools to take some of the lunar surface back onto the ship. This 100 kg of lunar cargo is moved back to the ship the same way stored in the containers brought to carry the 500 kg. Later, the astronauts can move these containers by hand to the command module. If nothing aberrant occurs, this cargo returns in the same way as the crew.

5.6 Propulsion & Orbital Maneuvering

5.6.1 Fuel

The liquid propellant NTO/MMH is chosen as the desired fuel for this mission based on the emphasis of reliability and technology readiness.

An appropriate orbital propulsion system is necessary after the launch vehicle raises the spacecraft to LEO. The propulsion system must have sufficient and controllable thrust for both the orbital maneuvering system (OMS) and reaction control system (RCS). Furthermore, it must be able to start

Propulsion	Example	T/W	Specific Impulse
Electric	Ion, resistojet, arcjet	10×10^{-6} – 10×10^{-2}	280–5000
Liquid Biprop / Solid	LOX/LH_2, SRB	10×10^{-2} – 100	200–464
Liquid Monoprop	N_2H_4, H_2O_2	10×10^{-2} – 10×10^{-1}	180–223
Units	—	—	s

Table 6: Basic propellant trade study indicates liquid bipropellant is best.

and stop quickly. After looking at the main propulsion types of today — nuclear, electric, solid, liquid monopropellant, and liquid bipropellant — liquid bipropellant was chosen. Nuclear rockets, whether by fission or fusion, might create fallout in the atmosphere if failure is to occur near Earth and development of a large nuclear engine has ceased since the 1960s. Electric rocket propulsion, though safe and efficient, produces too little thrust to be applicable in the timeframe required here. Solid propellants have preprogrammed burning processes and will not be of use in space. The simplicity of liquid monopropellants is overshadowed by the higher specific impulse of the bipropellants.

Currently, there are few bipropellant choices for RCS, notably NTO/MMH, NTO/-UDMH, and NTO/N_2H_4. Though there are developments underway with the fuels ethanol and liquid hydrogen, the TRL was judged to be too low. The main decision then is to find a liquid bipropellant for the OMS based on seven criteriaŮefficiency, safety, reliability, complexity, density, engine, and readiness — with reliability and readiness considered the most important. The tradeoff study indicated that NTO/MMH is the best propellant option with NTO/UDMH closely following.

There are several options for the storage of the liquid bipropellants. Since surface tension systems are the simplest and the EADS Space Transportation offers a wide variety of these propellant tanks, a logical choice is to use a surface tension propellant tank. The OST

Criteria	NTO MMH	NTO UDMH	NTO N_2H_4	LOX EtOH	LOX CH_4	LOX LH_2
Efficiency (Isp)	0.080	0.079	0.091	0.146	0.230	0.375
Safety	0.036	0.054	0.046	0.288	0.288	0.288
Reliability	0.321	0.321	0.227	0.051	0.046	0.034
Complexity (# propellants)	0.267	0.267	0.267	0.067	0.067	0.067
Density	0.219	0.191	0.361	0.121	0.086	0.022
Engine (Performance)	0.286	0.286	0.143	0.048	0.048	0.190
Readiness (TRL)	0.240	0.240	0.240	0.047	0.064	0.170

Table 7: NTO/MMH was judged to be the best from these seven criteria with emphasis on reliability and readiness.

22/X is especially attractive for the thrusters due to its variable volume from 70 0 − 1 108 L. It has a mass from 36 − 49 kg using pressure gas helium or nitrogen. The shape is Cassini domes on two sides with a variable cylinder length in the middle. The tank qualification is for a hypergolic system. However, there are some neccesary design changes for the tank since the tank is designed for 400 N of thrust. The propellant for each thruster is stored in each OST 22/X while 43 OST 22/Xs is needed for the main thruster, with 25 tanks needed to reach the moon and 18 to come back. This is assuming a rough estimate of height of 3 m and radius of 0.3 m for the tanks going there. For coming back, an estimate used height of 2 m and a radius 0.4 m is used.

5.6.2 Engines

The main engine is chosen to be the RS-72, a modified version of the Aestus 2 from the joint-collaboration of Rocketdyne and DaimerChrysler while the thruster chosen is the Aerojet 445.

The RS-72 was judged to be best in terms of T/W ratio and Isp though the thrust does have to be scaled up. RD-0242M is not chosen since it can only be used 6 times.

RS-72 has good thrust to weight ratio and decent specific impulse. Its vacuum Isp is 340 s, nozzle diameter is 1.3 m, overall length is 2.286 m, and nozzle area ratio is 300. The bipropellant turbopump engine has been scaled up to 990 000 N to provide more

Figure 14: Rocketdyne RS-72 engine.

Engine	T/W	Isp	Thrust
RS-72	41	340	55 400
Aestus	32	338	27 500
Aestus II	32	338	46 000
RS-23	1	313	26 680
Shuttle OME	23	316	26 700
RD-0242M	54	336	98 100
Units	—	s	N

Table 8: Artemis trade study highlighting the reasons for choosing EOR-Direct.

thrust since the current 55 400 N is too little thrust. Other engines looked into were not chosen because the RS-72 offered better performance. The Aestus 2 weighs more than the RS-72 but provides less thrust on top of a slightly lower Isp. Yangel's RD-8 was also considered but its thrust to weight ratio is half that of the RS-72 while its specific impulse is only a little better.

5.6.3 Mass Calculations

Total propellant on spacecraft to reach the moon is minimized with the usage of two turbopump engines, one of which is left on the moon for the return trip.

This mass does not include the propellant on the two EDS stages that propels the vehicle to low lunar orbit, hence the greater delta V for the return trip.

$$m_p = m_f \cdot \left(-1 + e^{\frac{\Delta V}{I_{sp} \cdot g}} \right) \qquad (1)$$

Calculations are done iteratively in the Propellant–Mass Calculator by first calculating the return trip's propellant mass using the rocket equation given by Eq. 1, assuming that all of the propellant in the first part of the trip has been exhausted. This mass includes the propellant mass needed for return trip and the propellant tank's weight. A margin of 2% is used excluding the 5% each for ullage and residual propellant volume. The final mass here is the spacecraft dry weight plus the 100 kg payload plus the 4 crew members' weight. Then the calculations are done again the same way but this time the propellant mass needed for return trip and its respective propellant tank is included in the final mass calculation in addition to the spacecraft dry weight, 500 kg payload, the human weight, weight of another engine, and pressurant weight. The latter is due to the RCS thrusters not being a turbopump and hence using a gas pressure system. Again, a 2% leeway is assumed. In the rocket equation, the gravitational constant g is assumed to be 9.81 g.

5.6.4 Interspace

In order to perform TLI and LOI burns, the EDS acts as the main propulsion device after docking in LEO. Two EDSs are docked in LEO with the LTS. Since the TLI EDS waits in LEO for 45 days and uses cryogenic fuel, it will boil off some of its fuel and therefore this lost fuel is accounted for when calculating the two EDS's wet masses.

In order to get the 72 mT LTS from LEO to LLO, the upper stage of the launch vehicle is required to perform TLI as well as complete LOI. The stage will also have to do a suborbital burn after the first stage separates in order to get into an assembly orbit of 56x 296 km at 28.5-deg. Such a stage, specially designed by the ESAS team for the CaLV used in Project Artemis, is referred to as the Earth Departure Stage (EDS).

The EDS is 22.7 m long and 8.38 m in diameter. It has a vacuum thrust capability of 1.22 MN and a vacuum Isp of 451.5 s at 100.0% power level. This conventional stage structure contains two J2-S+ engines, a thrust structure/boattail housing the engines, an Auxiliary Propulsion System (APS), and other stage subsystems. The EDS is configured with both an aft LOX tank and a 8.38 m diameter LH_2 tank which is connected to the LOX tank by an intertank structure. Figure 15 shows the structure of the EDS.

Unfortunately, the payload at LEO is too high for just one EDS, which at a maximum wet mass of 227 mT can get only 54.7 mT into LLO, to handle both TLI and the comple-

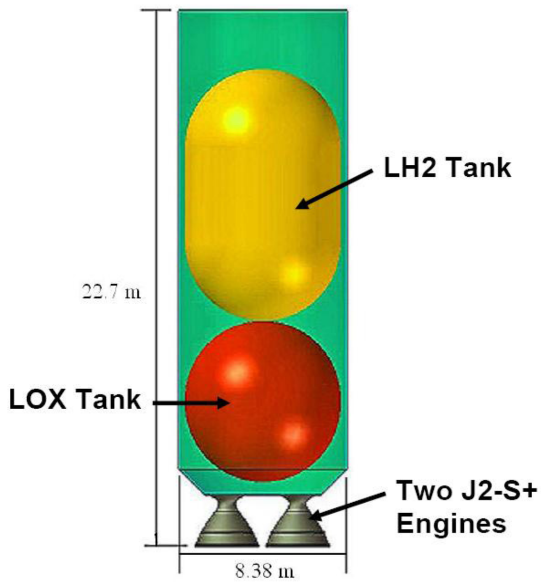


Figure 15: Earth Departure Stage

tion of LOI. Due to this fact, two EDSs are docked with the LTS in LEO. One EDS is launched up by the CaLV and waits at LEO for a period up to 45 days. It performs TLI and is launched at a wet mass of 216 mT. After burnout, the EDS's APS places it in a disposal solar orbit. The second EDS (which launches up with the LTS) takes over after the first EDS burns out and performs LOI in order to insert the LTS into LLO. It is launched at a wet mass of 217 mT.

The masses of the EDSs are calculated using the PMC just like the OMS and the RCS. For the LOI burn, a delta-v of $1078\,\mathrm{m{\cdot}s^{-1}}$ is used since this is the required delta-v to reach the far side of the moon. For TLI and suborbital burns, delta-vs of $3120\,\mathrm{m{\cdot}s^{-1}}$ and $2356\,\mathrm{m{\cdot}s^{-1}}$ respectively are used.

One of the cons of using cryogenic fuel is the fact that some fuel is lost due to the very low boiling temperatures. Therefore, the LOX and LH_2 tanks have a given boil off per day after launch. This primarily affects the TLI burning EDS's mass since it sits up in orbit up to 45 days. Team LETO uses very conservative estimates to determine that the EDS loses at most 13.5% of LH2 and 1.8% of LOX while waiting in LEO for the LTS and second EDS. Should the LTS and second EDS dock up with the first EDS before 45 days, any propellant not boiled and not burned off is expended with the jettisoned EDSs.

Though it is beyond the scope of Project Artemis to perform a trade study using other possible upper stages (since this would be developing a new LV), a study is done to find if is at all possible to use just one EDS in order to perform a suborbital burn with the LTS, perform TLI, and perform LOI. Table 9 shows the findings of this study. Designated as EDS3, such and EDS's mass is beyond the lifting capability of the CaLV. Therefore the aforementioned two EDSs are instead used

EDS Stages Flight Path	***Delta-V*** EDS 1	EDS 2	EDS 3
Suborbital Burn	2356	2356	2356
TLI	3120	N/A	N/A
LOI	N/A	1078	N/A
Both	N/A	N/A	4198
	EDS 1 + EDS 2		EDS3
Propellant Mass	433 678		304 679
Units	kg	kg	kg

Table 9: EDS Mass Trade Study.

when inserting the LTS from LEO to LLO.

5.7 Attitude Determination & Control

Attitude determination for the vehicle is controlled by thrusters. Aerojet's Aerojet 445 was selected due to a very good thrust to weight ratio of 24 and a decent specific impulse of 309.

The NTO/MMH bipropellant thruster with a mass of merely 2 kg provides 450 N of thrust. Thrusters are positioned in 4 groups of 4 thrusters each on the service module and 4 groups of 3 thrusters each on the command module. For the service module thrusters, each group can point in two directions whereas for the command module each group has thrusters pointed in x, y, and z axes to provide pitch, yaw, and roll moments. The thrusters and their respective thrust directions are represented by red dots and lines respectively. Blue means that the thruster is through the ship.

5.8 Guidance, Navigation, & Control

Project Artemis employs flight proven and highly reliable GNC system to ensure the success of the mission

Guidance, Navigation and Control is required during all three phases of the mission; the LEO rendezvous, during the translunar

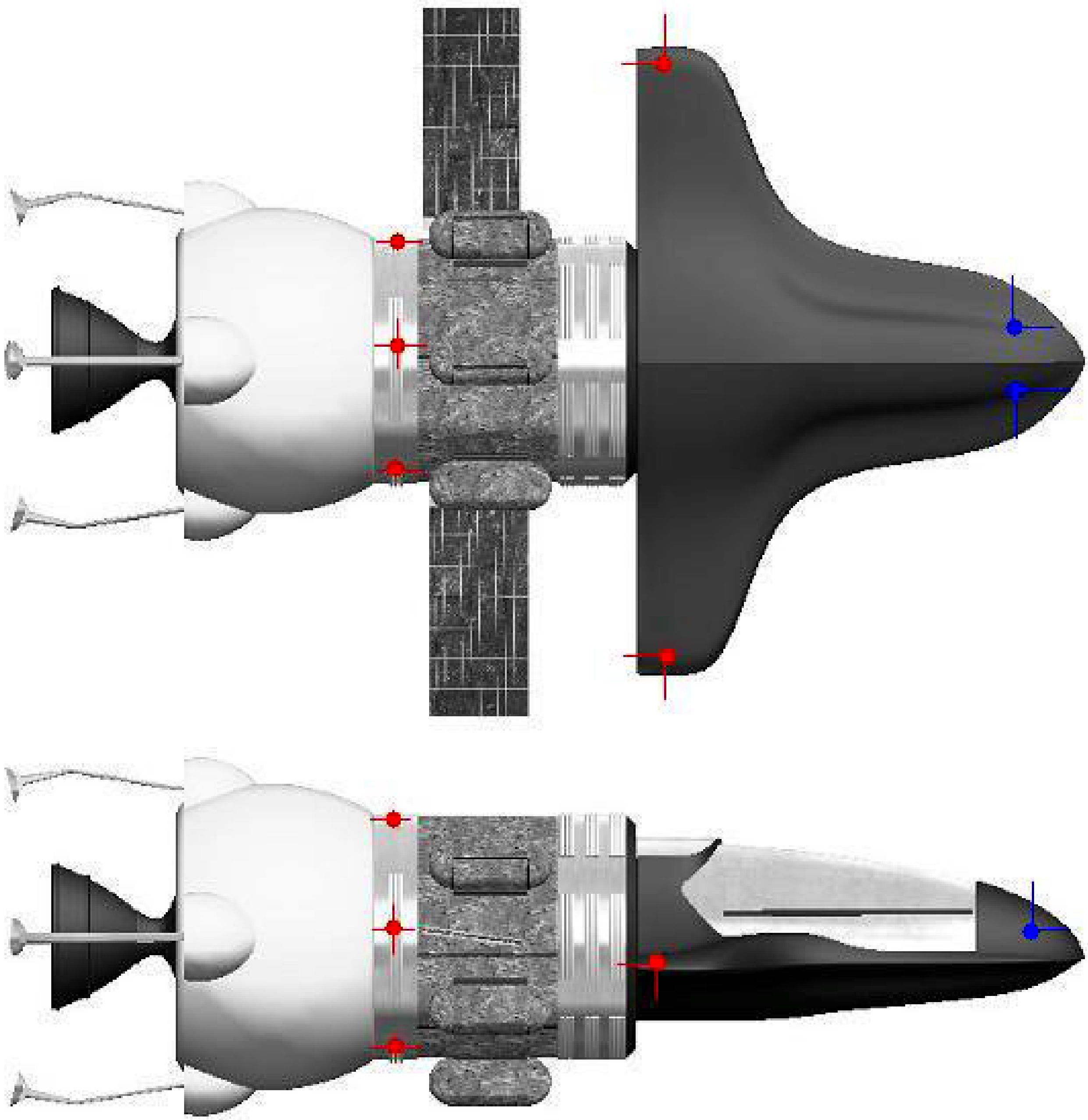


Figure 16: RCS setup with thruster locations and thrust directions indicated.

phase and during the lunar ascent and descent.

Different navigational techniques were examined and compared in detail before a selection was made. These techniques involved tracking via TDRSS, GPS, Space Sextant, Landmark tracking and Earth & Star sensing. The advantages and disadvantages of each option are listed in Table 10. TDRSS and GPS were eliminated because they can not be used on the moon. Space sextant is a great autonomous subsystem but too heavy and requires high power to maintain it. Thus, a star scanner was chosen to be used on Project Artemis for navigation.

Artemis also uses IMU manufactured by Northrop during the launch and the landing phase of the mission. The bias in the IMU will be regularly updated by the accurate readings from the star scanner. The table below shows the mass and power breakdown of the GNC subsystem.

5.9 Communications

The communication system will be composed of S Band transmitters and receivers in order to simplify the system with no negative aspects in safety or reliability.

The Communications Subsystem must provide a steady and reliable link between Ground Control and Project Artemis. The primary function of the link is to supply Data for the health and safety of the crew and vessel during the Launch, Docking, Journey to/from the lunar surface, and Re-entry; A secondary, but also very important aspect includes the supply of Video from the Lunar Surface to incite interest in Space Exploration and Science and encourage the continuous exploration of Space.

In order to perform this study, the bandwidth requirements of Project Artemis first had to be determined. Command and health and telemetry data is the most important aspect of a communications, as it is vital to the function of Project Artemis. Also important, however, is bandwidth allotment for science and video; it is vital in fueling interest in future missions to the moon and beyond. After analyzing all of the possibilities, Project Artemis includes a 5 Mbps video feed in the bandwidth allotment.

Due to the determined TRL boundary, a few of the newer and more interesting tech-

Navigation Options	Advantages	Disadvantages
TDRS Tracking	High Accuracy	Not Autonomous Limited Range
GPS	High Accuracy	Semi-Autonomous
Space Sextant	Fully Autonomous	High Power, Heavy Not Flight Tested
Landmark Tracking	Uses Observation Payload Sensor Data	Concept Stage Difficult
Earth & Star Sensing	Nearly Continous Attitude Determination	Cost Complexity

Table 10: Comparison of navigation options.

Navigation Options	Manufacturer	Mass	Power
IMU	Northrop	3.25	22
Star Sensor (2)	Ball	4.8	18
Momentum Wheels	Honeywell	20	80
Total	—	28.05	120
Units	—	kg	W

Table 11: Breakdown of navigation options by manufacturer.

Channels Needed	Data Rate	Data Rate
4	56 000	224
1	5 000 000	5000
1	4000	4
1	8000	8
1	50 000	50
Units	bps	Kbps

Table 12: Project Artemis' Communications subsystem Bandwidth Allotments.

nologies, such as optical communications systems, are eliminated. The communications systems already in place also took precedence, due to the fact that this reduces costs for Project Artemis and maintains a high level of efficiency in the communications subsystem. After analyzing systems already in place, the Deep Space Network is chosen because of its high capabilities and general under-use in manned spacecraft missions in the past few decades. Utilizing the DSN's sizable power and large Antennas, the communications system architecture in Project Artemis can be made with much smaller power usage and antenna size. The remaining band choices and subsystem specifics are carefully analyzed to determine the best performance and cost.

Each communications band is completely analyzed by constructing the entire subsystem with each band as its base. Then each resulting subsystem is judged according to its complexity, mass, power requirement, and compatibility with existing NASA systems. The most emphasis is put on compatibility with systems already in use, these aspects being integral in the construction of Project Artemis. After analyzing these many aspects, the s-band and x-band systems are further analyzed in trade-off studies; the end result is that an S-band based communications system is to be implemented in Project Artemis.

The brilliance in the S-band system is in its capabilities. It has increased bandwidth over the S-band system as well as a smaller antenna size; the mass requirement for the S-band system is one of the smallest of any of the systems. There is no need for a complex system with the S-band frequencies, all data can be streamed continuously down to the DSN ground stations using one of its two hemispherical antennas while in space, and singular parabolic antenna for communications while on the moon. The system will operate at a downlink frequency of 2.2175 GHz and uplink frequency of 2.0495 GHz, utilizing 8FSK modulation due to its low Signal-to-Noise Ratio Requirement, its simplicity, its low susceptibility to phase disturbances, and its good Bit Error Rate performance. The system will consist of two transponders for redundancy, along with two filters for redundancy.

Another possibility for Project Artemis' Communications Subsystem is a less power intensive UHF system for communications between the two components while in Earth Orbit and doubling as a comm. system for Astronauts performing EVA. The reason behind this is the simplicity of the system and low power and mass requirements for close-range and low bandwidth communications.

Project Artemis relies on no further communications system development by NASA, this decreases costs and improves efficiency and reliability; however, if NASA does construct any Lunar Communications infrastructure, Project Artemis is fully capable of using any new systems, because it uses the most favored communications band of NASA.

5.10 Telemetry, Tracking & Command

By using innovative technology in TT&C and vehicle health monitoring, Project Artemis improves the overall safety factor throughout the duration of the mission.

In the past, manned space missions have lacked sufficient technology to be self-maintained. All data collected by onboard sensors had to be sent to computers and personnel on the ground to be interpreted. Information pertaining to what the sensors indicated then had to be relayed back to the astronauts so that any anomalies onboard could be nullified. This could potentially be a waste of valuable time in the event of a critical error onboard. For this reason, Project Artemis employs the use of an innovative Integrated

Vehicle Health Monitoring System (IVHM).

The IVHM system is produced by an Australian company, CSIRO (Commonwealth Scientific and Industrial Research Organization). This system is composed of several layers of sensors through the vehicle that constantly provide feedback to an onboard computer. Each of these sensors is capable of processing speeds in excess of 100 MIPS.

Sensor data is interpreted by the computer and outputted to one of several systems. If a problem is detected with attitude or control, the computer can make the adjustments to keep the vehicle on course. If a problem requiring crew attention is detected, the crew is alerted via caution and warning lights on the control panel. In a critical situation, the time saved by having data interpreted onboard could be crucial to both mission and crew survivability. Telemetry is also simultaneously sent to ground stations for redundancy purposes.

Maximum power requirements for this IVHM system are less than 400 W. This power requirement stems from the sensors and the computer that will interpret the data.

By using this innovative, low power IVHM system, Project Artemis vastly improves crew safety over previous manned missions.

5.11 Command & Data Handling

As the means of monitoring and controlling all the other subsystems, the command and data handling system was chosen to provide the necessary computing power with the least power and weight requirement.

The Command and Data Handling subsystem is responsible for processing all of the data transmitted to Project Artemis as well as the data output by the other various subsystems and feeding the results to each subsystem for the proper function of the mission. It is the brain of Project Artemis, and must process copious amounts of data; therefore, it was very important to find high performance systems that required low power and mass.

The command and data handling subsystem is responsible for processing all data, and deciding what to with the data. It controls Guidance, Navigation, and Control, the Life Support Subsystem, the Propulsion System, Reaction Control System, and the Power

System of Project Artemis. With all of the data the C&DH system must process, it is necessary to find a very reliable and high performance system to implement in Project Artemis.

Though there are many different subsystems available, Project Artemis found three systems early in the design process that stood out; one for its extremely modular design, and two for their high performance and low power and mass requirements. Though modularity is an important aspect of Project Artemis, the higher performance systems supply the raw processing power needed for the space mission.

The radiation shielding is also an important aspect in any Command and Data Handling system; while the numbers for radiation shielding are included for the both the Proton 100K and Proton 200K, no numbers are supplied by General Dynamics Advanced Information Systems regarding the radiation shielding of the Integrated Spacecraft Control system. In the end, it came down to the two systems designed and produced by Space Micro Inc., so further trade-off studies were performed.

In the end, the Proton 100K is chosen because of its higher TRL and comparable performance, along with the radiation shielding provided.

Project Artemis will employ three of the Proton 100K computers for primary use, with a fourth left as back-up. This will provide enough processing power and redundancy to meet the most extreme needs of Project Artemis' Command and Data Handling Subsystem. Each will have an individual monitor and keyboard/mouse, so each will be accessible to the crew of Project Artemis at anytime during the mission.

Because of the needs for Project Artemis, a power and mass efficient system is mandatory, and high performance is a must. This leads to the use of the Proton 100K-based Command and Data Handling Subsystem, an answer to the most extreme cases where processing power is needed most. In all of the studies, the foremost aspect of Project Artemis is Safety and Reliability, features provided by the amazing speed and performance of the Proton 100K.

5.12 Structural & Mechanical Systems

The structural subsystem of the LTS serves to carry the crew, support the payload, provide attachment points for internal components, protect against micrometeorite strikes, and maintain the overall shape and function of the spacecraft.

The LTV has been designed to withstand maximum launch and landing forces of 10 g's, with axial and lateral vibration dampening tuned to the selected launch vehicle.

A comparison of critical spacecraft material characteristics is presented in Table 13. Using this information, Team LETO has concluded that the majority of the internal structure for the Artemis craft will be comprised of either 6061 T6 or 7075 T6 aluminum alloys. Aluminum was selected for its high strength to density ratio, corrosion resistance, non-ferrous characteristics, ease of machining, and relatively low cost. For components requiring high-strength and stiffness or low-temperature survivability, the titanium alloy Ti_6Al_4V is used.

Landing struts, one of the most critical structural elements of the spacecraft, are composed of an aluminum 6061 T6 alloy manufactured into a crushable honeycomb structure that absorbs the force of impact on the lunar surface, thereby minimizing structural loading on the craft. The dynamic nature of this honeycomb configuration also reduces loading during the launch stage of the mission. This approach to minimizing structural loading on impact was employed by the Apollo spacecraft and has been proven to be both successful and reliable.

Additionally, all moving mechanical parts are adequately sealed to prevent contamination by lunar dust.

5.12.1 Radiation Shielding

The Artemis team recognizes the need for radiation protection for lunar missions to protect humans from Solar Particle Events (SPE's) for a safer and more reliable mission using material protection.

Team LETO's main concern is the safety of the crew and mission success as well. To protect the crew but keep a viable mission, the ALARA (As Low As Reasonably Achievable) principle was implemented due to high mass costs from having complete protection. The National Council on Radiation Protec-

Material	Density ρ	Young's Modulus E	Yield Strength S_y	$\frac{E}{\rho}$	$\frac{E^{-1/2}}{\rho}$	$\frac{E^{-1/3}}{\rho}$	$\frac{S_y}{\rho}$
Aluminum Alloys							
6061 T6	2800	68	276	24	2.9	1.5	98.6
7075 T6	2700	71	503	26	3.1	1.5	186.3
Magnesium Alloy							
AX31B	1700	45	220	26	3.9	2.1	129.4
Titanium Alloy							
Ti_6Al_4V	4400	110	825	25	2.4	1.1	187.5
Beryllium Alloy							
S 65 A	2000	304	207	151	8.7	3.4	103.5
Ferrous Alloys							
AM 350	7700	200	1034	26	1.84	0.8	134.4
304L Ann	7800	193	170	25	1.8	0.7	21.8
Units	$kg{\cdot}m^{-3}$	GPa	MPa	—	—	—	—

Table 13: Structural Material Properties

tion and Measurements (NCRP) limits were also considered. The values per unit of duration are given in Table 14 and the values per age and gender group are given in Table 15.

The architecture for the radiation shield protects the astronauts at all body parts, especially the torso, and is placed around the habitation module. The habitation module was chosen to decrease the mass of the CM.

Duration	Skin	Eye	BFO
Career	6.00	4.00	*Table 15*
Annual	3.00	2.00	0.50
30-Day	1.50	1.00	0.25
Units	Gy	Gy	Gy

Table 14: NCRP Allowable Limits

Person	25 yr	35 yr	45 yr	55 yr
Male	0.7	1.0	1.5	4.0
Female	0.4	0.6	0.9	3.0
Units	Gy	Gy	Gy	Gy

Table 15: NCRP Allowable Limits by Gender and Age

This architecture does not pose a threat to the astronauts because the major radiation events would be felt after departure from LEO. In the event that an SPE is detected, the astronauts will transfer to the SM for radiation protection. This architecture protects the majority of the sensitive organs in the body. To find the mass of the subsystem, Team LETO designed for a large SPE event on September 1984. Protecting for twice the flare intensity of the baseline SPE would give protection of about 90% of the worst case scenario. Artemis decided that protecting for the worst case scenario was not necessary since SPE events of that intensity occur only once or twice per century.

The kind of material chosen for radiation protection was a combination of aluminum from the spacecraft and polyethylene. Polyethylene was chosen because it adds to the aluminum structural protection as well as being lightweight. Also, in terms of other plastics polyethylene is the best protector.

The maximum radiation exposures that the crew is expected to absorb is 30 cGy. This number was found by setting the thick area numbers of about 9 g$\cdot$cm^{-2}. It must be noted that these are numbers for maximum exposure, not expected exposure, which is considerably less. The expected exposures are in the milli-Gray range. If the radiation shielding were equally distributed along the service module, it would come out to a thickness of about 4.5 g$\cdot$cm^{-2}, which gives 100 cGy of absorbance. All these numbers are still well within the limits placed by NCRP.

5.13 Thermal Protection

The Thermal Protection System will be composed of ARMOR, RCC, AFRSI and FRSI on the command module based on the temperatures of another winged bodied spacecraft: the Space Shuttle.

The thermal protection system (TPS) is a very important component of the spacecraft due to its role of making sure that the ship will be able to survive the high temperature increases during the re-entry phase. The TPS must resist temperatures up to 1600 C, have resistance to water, and have ease of maintenance and repair. There are many different materials that can be considered for the TPS, each with their own distinct pros and cons.

Ceramic Tiles are made up of low density, high-purity silica. High-temperature Reusable Surface Insulation (HRSI), having a maximum temperature protection of 1260 C, and the Low-temperature Reusable Surface Insulation (LRSI), having a maximum temperature protection of 650 C, have been used in the Space Shuttle program with success. However, the ceramic tiles are not waterproof, which makes launching during harsh weather conditions impossible. Ceramic tiles also chips very easily during launch and reentry and have caused major problems in the past. They also take a great deal of time to fix, increasing the turn around time of the vehicle.

Reinforced Carbon Carbon (RCC) tiles are produced by impregnating a graphitized rayon cloth with a phenolic resin. It is then cured and pyrolized to convert the resin to carbon. Then, it is impregnated with furfural alcohol, then cured and pyrolized again for the conversion to carbon. This process is repeated until the desired properties are achieved. RCC has an operating range of $-16\,0 - 1\,650$ C, which is about the same range as the temperature variance of the ship during its mission. RCC is also light-weight and rugged, making it an ideal TPS. However, also due to its design process, RCC is very expansive to produce, and must be used sparingly.

Blankets have a lower temperature protection, but save on cost and mass. Advanced Flexible Reusable Surface Insulation Blankets (AFRSI) are made from high purity silica and amorphous silica fibers which are woven between high temperature silica fabric and low temperature glass fabric. ARFSI have the same potential as the LRSI, but weighs drastically less. AFRSI are more durable, and are quicker to fabricate, install and maintain than the tiles.

A new metallic TPS has been produced and been tested to replace the tiles. The new Adaptable, Robust, Metallic, Opera-

TPS	Temp Range	Weight	Reliability	Cost	TRL
RCC	-160–1650	3.9-5.8	5	1	9
HRSI	-120–1260	10.3-12.9	3	3	9
LRSI	-106–650	7.8-8.4	3	3	9
AFRSI	-106–650	4.8-7.3	4	4	9
FRSI	-106–371	3.2-4	4	5	9
ARMOR	-120–1100 (1650)	7.8-9.7	4	2(4)	7(9)
Units	C	kg·m^{-2}	1–5	1–5	—

Table 16: Shows the properties of the different TPS to be considered for the spacecraft.

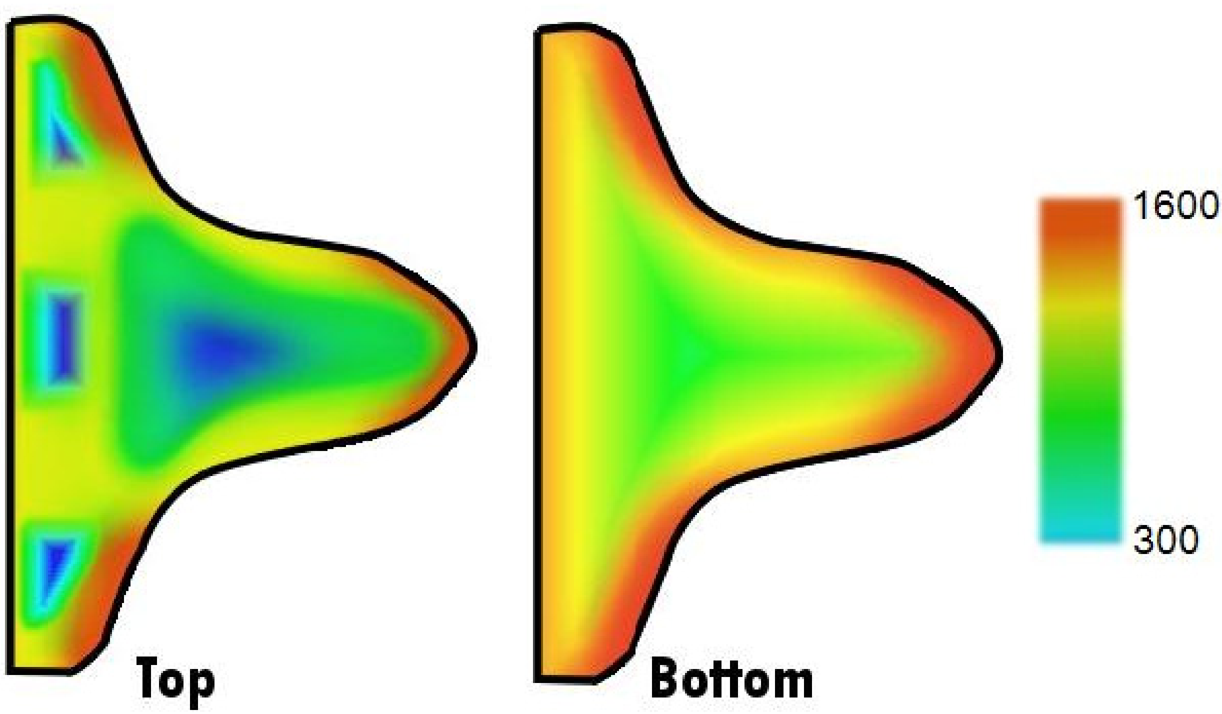


Figure 17: Shows the estimated temperature variance of the spacecraft's body.

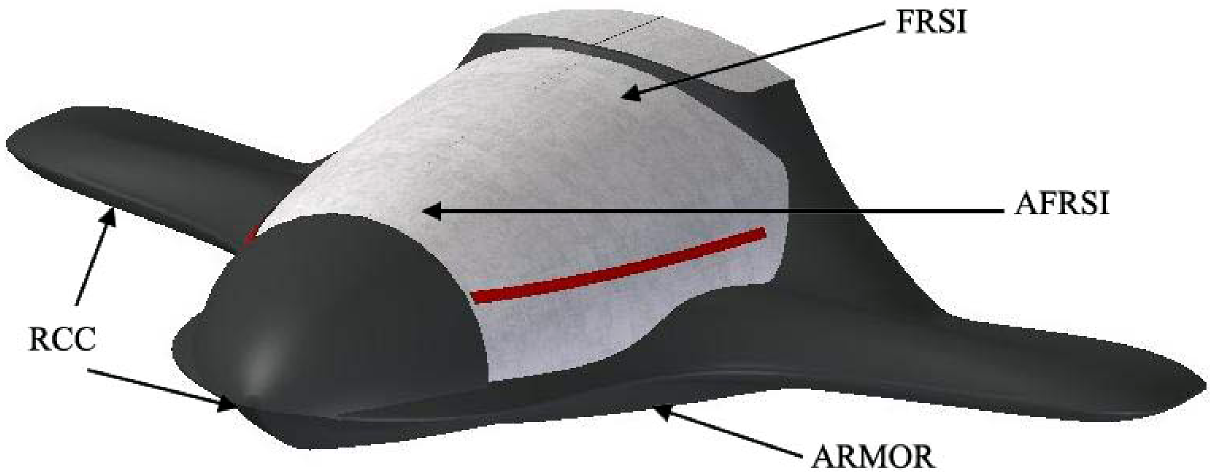


Figure 18: Shows the TPS materials used on the various parts of the command module.

ble, Reusable (ARMOR) TPS is made up of inconel 617 in a honeycomb structure. Its performance is the same as HRSI but weighs less. Also, it is designed to resist water, eliminating time consuming water-proofing before each mission. It is also attached by mechanical fasteners, which makes installing and replacing the ARMOR quick and simple. ARMOR is still in development, and it has the potential to have a temperature protection as high as the RCC. By the production stage, the ARMOR protection should be higher than the HRSI tiles and come close to the maximum temperature protection of the RCC.

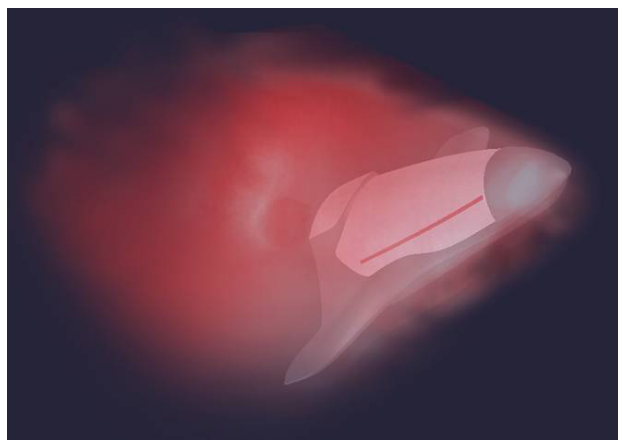


Figure 19: Adequate TPS has been provided for the safe reentry of the vehicle.

The temperature range during re-entry is different for each section of the spacecraft, and can be compared to the Space Shuttle's winged body shape. From the temperature calculations of the Space Shuttle, it can be seen that the temperature of the nose of the ship and the leading edge of the wings are too high to use anything except RCC. ARMOR is used for the underside of the ship, replacing the HRSI tiles used by the Space Shuttle. The top portion of the spacecraft, where temperatures are relatively cooler, a mixture of AFRSI and FRSI are used, replacing the LRSI tiles.

Before launch, the AFRSI and FRSI blankets must be treated with a water resistant substance to protect it from water. The ARMOR will not need to be treated, saving time before launch. Also, using ARMOR for the underside of the spacecraft, where damage is most sustained, will save a great deal of time for repairs since it is easily removed and attached, which will greatly decrease the turn around time for the spacecraft.

The Thermal Protection System for the spacecraft was picked with the criteria of being the safest and most reliable option, each component being the ideal selection for its part on the spacecraft. The choices make the TPS the lightest solution possible, weighing in at 1159.4 kg, with no sacrifice on its performance or reliability.

5.14 Power

Project Artemis incorporates safe, lightweight, and reliable energy storage and production into the CEV design through the use of solar panels and lithium ion batteries.

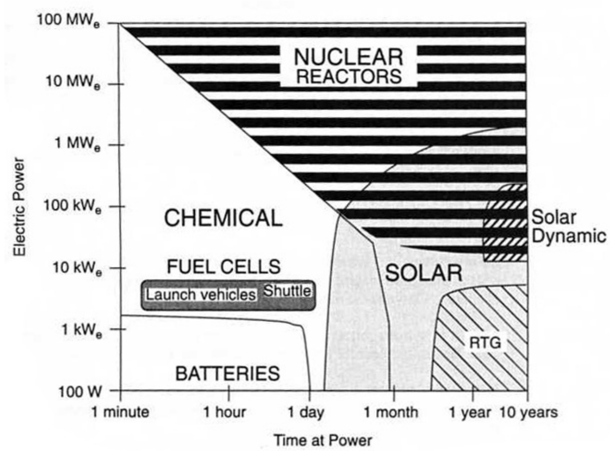


Figure 20: Suggested power systems as a function of mission duration and total power requirements.

Some of the driving factors for the development of the CEV power source were the necessity to provide continuous power without occupying a large volume or mass. In addition, the power requirement for Project Artemis was only on the scale of several kilowatts for a duration of two weeks, which corresponds to a suggested power system of solar panels from Figure 20. Due to this restriction, Project Artemis decided to pursue the use of solar panels as the primary long term, lightweight power source.

Ultra-Triple Junction Gallium Arsenide Solar Panels, produced by Spectrolab, were chosen to provide power throughout the majority of the mission. These were chosen pri-

Solar Panel	Specific Power	Specific Mass	Efficiency %
Gallium-Arsenide			
Single Junction	241	1.89	18
Dual Junction	266	1.89	19
Triple Junction	302	2.36	22
Improved Triple Junction	330	2.36	24
Ultra Triple Junction	350	2.06	26
Indium Phosphide	246.6	-	18
Amorphous Silicon	68.5	1.79	5
Silicon	191.8	1.64	14
Units	$W \cdot m^{-2}$	$kg \cdot m^{-2}$	—

Table 17: Comparison of solar panel parameters.

marily based on their efficiency and reliability factors. These panels have been extensively flight tested and proven, and thus have a TRL level of 9. Due to their higher efficiency, the UTJ GaAs panels have a power density of around $350\,W \cdot m^{-2}$, which is much greater than any of the other options considered. The specific mass of the panels was not the best of all the panels considered, however, they remained the best option.

The panels have been sized to provide adequate power for the system during peak power requirements. This peak power requirement is around 6000 W. In order to achieve this power requirement, the panels are sized at $17.1\,m^2$. This is evenly divided between two panels on either side of the service module. These twin panels are each located on booms away from the service module, this is to allow a maximum amount of solar energy to reach the surface of the panels. This also allows the panels to be articulated to remain perpendicular to the sun at all times. This is important due to the fact that the amount of solar energy collected falls off as a cosine of the angle between the panels and the sun.

During launch, the solar panels will be re-

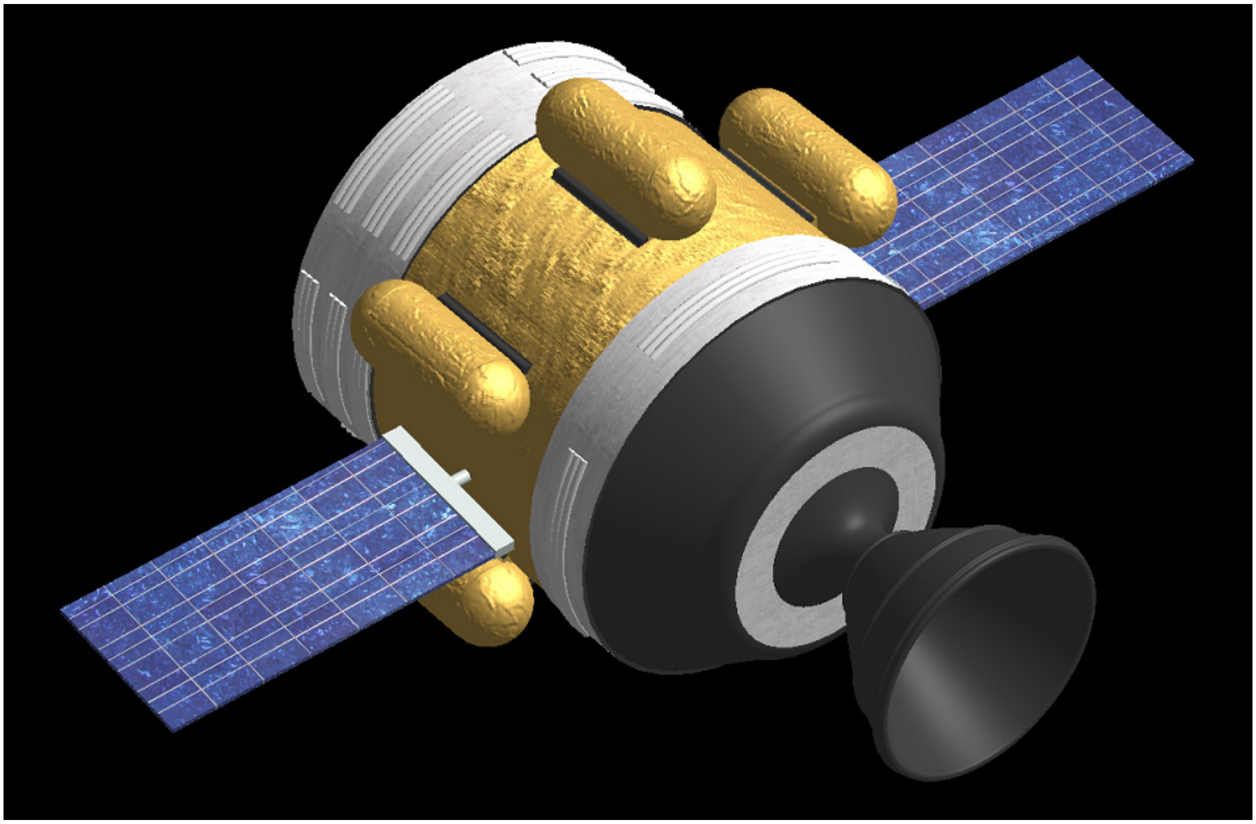


Figure 21: Project Artemis Service Module with solar panels extended.

tracted into the vehicle. The system will run on batteries for the seven minutes of launch. As soon as LEO has been established, the panels will be extended and energy production will begin.

The panels will remain in the extended position until the final lunar decent. At this time, they will once again be retracted into the body of the vehicle and the system will run on battery power. This is done in an attempt to mitigate the effects of lunar dust, expelled upward by the engine plume of the descending vehicle, from settling on the surface of the panels, as this would cause their efficiency to drop below an acceptable level. Soon after landing, the panels will once again be extended, and they will provide power for the entire lunar surface duration.

Since Project Artemis utilizes solar panels as the primary power source, a form of energy storage is necessary to supply power to the CEV during critical mission phases when the solar panels are not active and for periods of solar eclipse when the panels are unable to function.

Fuel Cell & Batteries Team LETO examined two types of feasible energy storage: regenerative fuel cells and batteries.

The primary limiting factor for the use of regenerative fuel cells is the mass of the liquid hydrogen and oxygen fuel. The mass of the liquid hydrogen and oxygen is on the range 100 kg up to 1000 kg, depending on mission duration. The substantially larger mass than batteries, along with the relatively low TRL (around 4), discouraged Team LETO from pursuing the use of regenerative fuel cells.

For secondary batteries there are generally three types that have been historically used and are currently used in spacecraft. Nickel Cadmium (NiCd) batteries are a rather outdated form of energy storage, with low specific-energy and low energy density. Nickel hydrogen (NiH_2) batteries are only slightly better than NiCd batteries in terms of specific energy and energy density; however, NiH_2 batteries have been widely used in satellites, making them a flight worthy component. Lithium ion (Li-ion) batteries are an extremely advanced battery type since they have been developed for use in the consumer market (cellular phone batteries and laptop computers). Li-ion batteries have high specific energies (greater than $100\,\mathrm{W{\cdot}h{\cdot}kg^{-1}}$), making them lighter than traditional secondary batteries. Li-ion batteries also have a relatively high energy density on the magnitude of $32\,\mathrm{W{\cdot}h{\cdot}kg^{-1}}$.

The lithium ion batteries that Project Artemis utilizes were selected because of the high specific-energy and high energy-density compared to other types of batteries. Four batteries produced by Mitsubishi Electric provide the energy storage for the mission duration. The four batteries are located in the rear of the command module, and each has a capacity of around $8460\,\mathrm{W{\cdot}h}$ of energy. The mass of each battery is 81.0 kg with the total mass of all four batteries coming to 324 kg. Dividing the energy of the batteries by the mass, the specific energy comes to $106.7\,\mathrm{W{\cdot}h{\cdot}kg^{-1}}$, which is substantially higher than NiCd and NiH_2, whose specific energies are on the magnitude of $40-50\,\mathrm{W{\cdot}h{\cdot}kg^{-1}}$ respectively.

Three of the four batteries on the command module will be used as energy storage,

Battery	*Battery* Specific Energy	*Battery* Energy Density	*Cell* Specific Energy	*Cell* Energy Density
NiCd	35	45	39	137
NiH_2	49	25	60–80	64
NiMH	60	86	—	—
Li-ion	85–175	160	100–200	260
NaS	132	165	110	200
Units	$W{\cdot}h{\cdot}kg^{-1}$	$W{\cdot}h{\cdot}L^{-1}$	$W{\cdot}h{\cdot}kg^{-1}$	$W{\cdot}h{\cdot}L^{-1}$

Table 18: Comparison of various battery parameters.

and the fourth battery will be used for backup and will provide redundancy to the system. With the peak power of the CEV totaling around 6000 W of power, the three batteries can provide around four hours of power at peak demand. The power by the batteries will be primarily used during eclipse, launch, lunar landing, and reentry as these are all periods when the solar panels are not functioning.

Power Management & Distribution Power management and distribution, commonly known as PMAD, is another important element of Project Artemis' power system. PMAD is responsible for regulating the voltage from the solar panels and for cleaning and distributing the current produced by the solar panels. PMAD also regulates the recharging of the batteries and monitors the power sent out to the various components of the spacecraft. PMAD makes up a significant portion of the power system mass at around 197 kg.

Through the use of the lightweight UTJ GaAs solar panels and the lightweight Li-ion batteries, Project Artemis minimizes system mass, without sacrificing reliability and system performance.

6 Architectural Elements

6.1 Launch Vehicle

In order to meet the boundaries that the RFP and Team Artemis set for the LV, the five SSME core w/ five RSRBs CaLV is determined to be the best LV for the LTS. The CaLV is one of the few LVs that can fit the payload requirements of Project LETO and launch the desired payload up to LEO in less than four launches. It is the safest LV and among the most reliable LVs as well, which are the chief factors that Team Artemis uses to determine the best subsystem as stated in the Design Methodology.

6.1.1 Requirements

According to the RFP, the launch vehicle is strictly to define the LEO altitude and inclination as well as the LTS mass constraints. Other requirements, which are used directly to constrain the LV, include the ability to launch a LTS carrying 500 kg of payload up to LEO, have a minimum ability to launch up one mission per year, and follow the Vision for Space Exploration deadline of a mission back to the moon by 2020. Finally, the RFP required that a trade study be done by current U.S. and International Expendable Launch Vehicles (ELVs).

Here Team LETO wants to make it clear that since no LV as of today has the payload capacity to lift the Project LETO LTS without a minimum of four launches (which according to the ESAS team leads to a very unfavorable LOM), the Vision for Space Exploration deadline of 2015–2020 for an existing lunar LV is used to help define "current LVs." It is beyond the scope of Project Artemis to create a new launch system and existing launch families such as the Atlas V or Delta IV must be used in conjunction with the 2015–2020 period in order to define a "current LV". Therefore any LV examined fits along with the RFP. Furthermore, Team LETO sets its own requirements that the LV be firstly safe, and secondly that the LV be reliable. Since the LV is not actually part of the LTS, reusability is seen as a feature not necessary in deciding the best LV for Project Artemis.

6.1.2 Survey of Existing Launch Systems

Due to the payload requirements, several heavy lift launch vehicles are examined by

Team LETO as shown by Table 19. All of these LVs stem from an existing family and fit the definition of a "current LV." Many of these options can be eliminated because of their relatively low payload capacity. Since these LVs would require four plus launches to send the desired payload into orbit, they adversely affect the reliability of Project Artemis as stated earlier in the requirements. This leaves three options that fit the necessary payload capability: a shuttle derivative and two Atlas V derivatives (all developed by the ESAS team for the study of the best lunar mission LV).

Launch Vehicles	Payload to LEO	Fairing Size	Human-Rated	LOM	LOC	Cost per Launch	DDT&E
Man-rated Delta IV Heavy	28	19.1 x 5.1	Yes	1 in 172	1 in 1110	$267	$8,155
Atlas V Heavy Phase 3A	93.8	23.4 x 5.4	Yes	1 in 88	1 in 612	$496	$7,325
Atlas V Heavy Phase X	95	23.4 x 8	Yes	1 in 71	1 in 536	$451	$16,016
CaLV	106 (126 w/EDS)	22 x 8.4	Yes	1 in 124	1 in 2021	$417	$12,416
Falcon 9	24.75	15 x 5.2	No	N/A	N/A	$101	$649
Angara A5/UHOM	28.5	19.6 x 5.1	No	N/A	N/A	$124	$2,677
Units	mT	m	—	—	—	FY'18 Millions	FY'18 Millions

Table 19: Launch Vehicle Trade Study.

Shuttle Derived LV Known as the Cargo Launch Vehicle (CaLV), this Shuttle derived LV uses the same Reusable Solid Rocket Motors (RSRMs) that the current Space Shuttle LV uses for boosters. These RSRMs are stacked five high on two booster stages. The core stage is comprised of 5 RS-25 SSMEs. The CaLV features a 8.38 m diameter fairing and stands at 109 m tall. It launches at Kennedy Space Center, Pad 39.

At sea level conditions, the core stage has an engine thrust of 1.67 MN and an engine Isp of 361.3 s. Under ambient vacuum conditions, the engine thrust and engine Isp are 2.09 MN and 452.1 s respectively. The upper stage or the EDS is comprised of two J2-S+ engines which deliver 1.22 MN of thrust at vacuum conditions.

Atlas V Derived LVs Using man-rated RD-180 main engines as well as RD-180 booster stages, the Atlas V Phase 3A and Phase X LVs can deliver from 94 – 95 mT into LEO. The Phase 3A has a 5 m diameter fairing. The wet mass of the Phase 3A is 2823 mT with a liftoff T/W ratio of 1.39 g. The evolved Atlas V, known as the Phase X features two Atlas V boosters and has a diameter of 8 m and has a wet mass of 2270 mT. However, it only has a liftoff T/W of only 1.21 g. The Atlas V derivatives are launched from Space Launch Complex 41 at Cape Canaveral Air Force Station.

6.1.3 Performance and Readiness

As Table 19 shows, the CaLV has by far the best LOM and LOC ratios than either of the two Atlas V derivatives. Since these ratios correlate with the reliability and safety of the

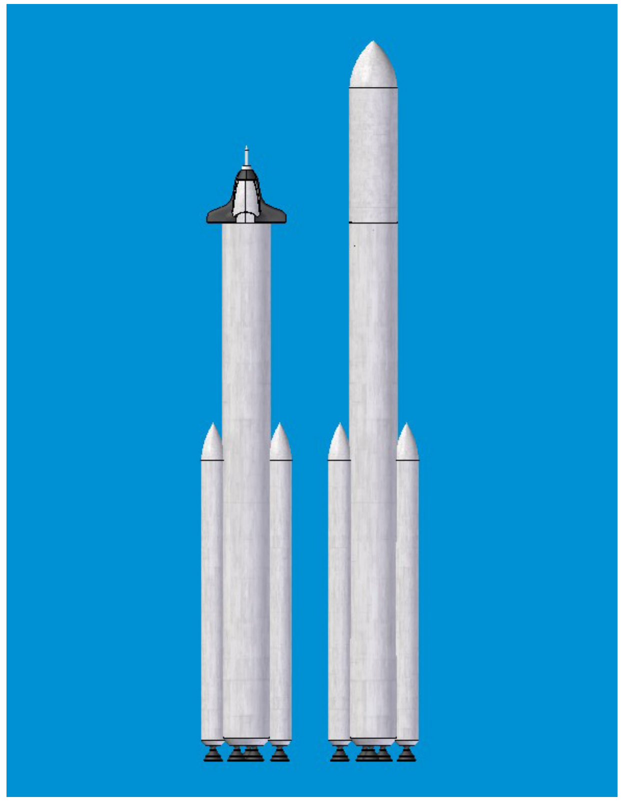


Figure 22: Cargo Launch Vehicle (CaLV)

CaLV, it is Team LETO's choice for Project Artemis. Figure 22 shows the CaLV with the LTS mounted on top and the fixed winged body of the CM sticking out of the shroud as well as a CaLV shown with the missing fairing in place when the EDS launches by itself.

The CaLV will have no problem fitting the EDS with the LTS since the LM used is derived from the Apollo missions and therefore comparable to the LSAM that ESAS designed the CaLV to attach to. However, the CaLV will have a problem launching the winged body of the CM which will exert a horizontal "lifting" force. Therefore, the engines must be gimbaled on the CaLV when launching the LTS in order to counteract this force.

Since all of the subsystems of the CaLV have been tested in a relevant environment and also feature a great deal of heritage, the CaLV receives a TRL of 6. However, according to the ESAS team, the CaLV is ready for launch by 2018 which is in the required realm of 2015–2020. The CaLV also has the ability to launch the LTS and two EDSs more than once a year. It can carry a payload of 125 mT into LEO (106 mT without the EDS) and insert the EDS as well as the payload into 56x 296 km at 28.5-deg.

6.2 Abort System

Project Artemis ensures crew safety and survivability by providing abort planning and systems during launch and throughout the mission duration.

The abort system for Project Artemis consists of two separate systems, the launch escape system and the abort guidance system. The launch escape system (LES) provides a mission abort during launch, while the abort guidance system (AGS) provides a redundant navigation and guidance system in case of a primary system failure.

The launch abort system is designed for three situations: an immediate launch abort within one minute of rocket ignition, an abort after the LES has routinely been jettisoned, and an orbital abort.

For an immediate launch abort, the crew would make use of the launch escape system (LES). The LES is similar in design to those used for the Apollo and Soyuz missions. The LES is essentially a small rocket motor assembly which is attached to the command module. In case of a catastrophic failure shortly after launch, the rocket engines on the LES would ignite pulling the command module to an altitude of around 3000 m. At this altitude, the LES would be jettisoned along with any remaining rocket fuel and the command module would parachute back to the ground.

For an abort that occurs after the LES is jettisoned, the command module would separate from the lunar module and the launch vehicle adapter, and will use the primary engine of the lunar module to further separate from the launch vehicle. After safe separation from the launch vehicle, the command module would separate from the lunar module and service module and then perform an emergency reentry.

For an orbital abort, the CEV would make use of the EDS stages to obtain a stable Earth orbit, from where the ground crew would determine when the command module should reenter the atmosphere.

For crew safety and redundancy after launch, the Artemis CEV has a redundant secondary navigation and guidance system called the abort guidance and navigation system (AGS). The AGS would ensure crew safety during cruise to and from the moon in case of a primary navigation and guidance system (PNGS) failure.

The AGS is an independent navigational and guidance unit on the CEV. In the event of a primary guidance system failure, the AGS is able to take control of the primary navigational functions along with both the ascent and descent stages of the Artemis CEV.

In the event of a mission abort during cruise to the moon, the AGS or PNGS would make use of the free return orbit to swing the CEV back to Earth.

For an abort during the lunar landing phase of the mission, the AGS or PNGS would jettison the lunar module and make use of the ascent engine to put the CEV on a direct return course to the Earth.

The abort systems of Project Artemis combined with contingency planning ensures crew survivability and safety throughout the mission.

6.3 Earth Landing & Recovery

A winged body shape with a parafoil was used for Earth landing to provide the stability and control to land at any desired location with safety and reliability at its max.

As requested by the RFP, the Earth Landing System is to land a crew of four and 100 kg of cargo safely upon the mission's return to the Earth. In, addition, it is necessary for the Earth Landing System (ELS) to land the CM at a predestined location on Earth. Another constraint for the ELS is that it is to be reusable within one year of use for the next mission. Therefore, safety, payload weight, reusability, and precision steering were used as the main criteria during the trade studies for the ELS.

The options considered for the ELS in-

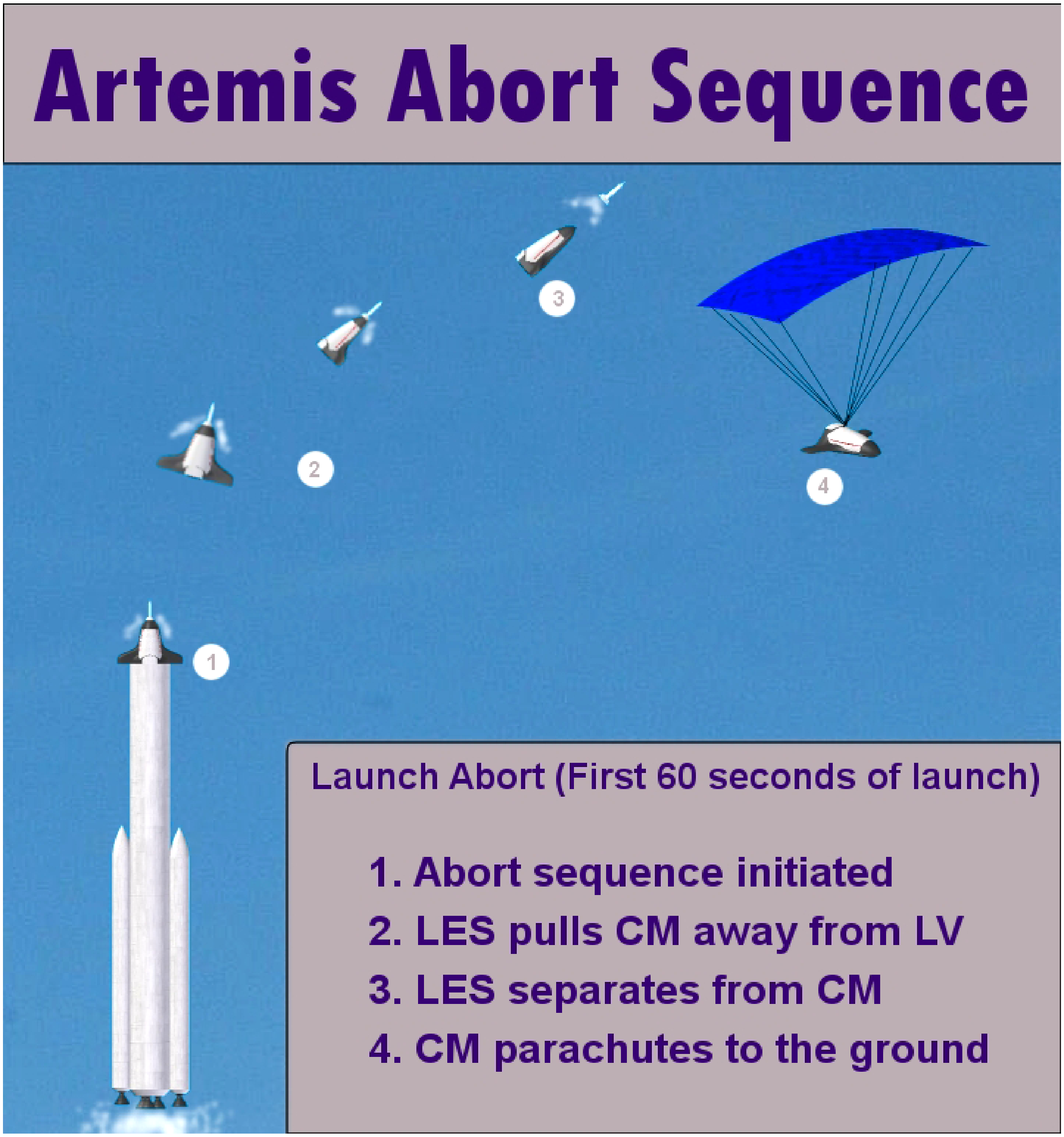


Figure 23: Project Artemis Service abort sequence for an immediate launch abort.

Landing Options	Advantages	Disadvantages
Parafoil	Steering Flared Braking Developed by NASA for X-38 Advanced Guidance System Reliable	Maximum Payload of 10 900 kg
Parachute	Reliable Used for Apollo Low Landing Speed	No Steering High Mass
Winged Body	Used on Space Shuttle No External Braking System Short Turnaround Time Extremely Reliable	High Landing Speed High Mass

Table 20: Comparison of landing options.

cluded an air-ram parafoil, regular parachute, and a winged body design. While the winged body design was the most reliable by having no external parts, and the most reusable option, the wing-mass to payload-mass ratio became ineffective for a space mission. Since the CM is much smaller than the Space Shuttle it cannot afford to have the extra mass from large wings. A winged-body landing would also require a 700 kg tricycle retractable landing-gear system due to the high landing speeds, further increasing the mass.

Next considered was a parachute system comparable to the Apollo mission's parachute landing; however, parachutes do not provide steering for a precision landing and would also have to make an ocean landing in order to be safe. Ocean landings were ruled out because of the decreased reusability of the CM in water landings.

The last option was a ram-air parafoil landing system. This 700 m^2. parafoil was designed by NASA for the X-38 ISS crew return vehicle. It provides the steering capabilities necessary for the landing. The parafoil also provides a much slower, therefore safer, landing by flaring the parafoil just before landing. This option allows for a safe skid landing in the desert. This parachute, however, has a maximum payload of 10 886 kg and by itself cannot safely return the crew and cargo to earth.

Since none of the options alone met the requirements of the ELS, it was decided that a parafoil and winged-body combination became the best choice. The parafoil eliminates the high landing speeds and wing-mass, while the winged-body reduces the payload-mass by providing lift, the negative qualities of both options are removed in their combination.

The parafoil is a 700 m^2 wing-shaped parachute, very similar to parachutes used by skydivers. The parafoil was initially designed by NASA for the X-38 crew return vehicle. Although the X-38 program was cancelled due to high cost, the parafoil was fully developed and was drop-tested 13 times at the 700 m^2 size, giving it a TRL of 8.

The parafoil is deployed in stages. The first stage is a 2.7 m diameter pilot parachute deployed by a mortar at an altitude 10 000 m. While beginning to slow down the CM, the pilot chute also orients the CM onto the correct axis for the deployment of the next stage.

The next stage of parafoil deployment is the 30.5 m diameter drogue-chute. The drogue-chute slows the CM from a velocity of Mach 0.8 to Mach 0.25. A maximum of 3 g's are experienced during the drogue-chute extraction. The drogue-chute is also used to extract the parafoil at the desired velocity.

Due to the large size of the parafoil, it is extracted in segments to prevent the tangling of wires and to minimize forces exerted on the parachute. The parafoil is divided into 32 cells, these cells are inflated starting at the center cells moving outwards. The drogue is attached to the parafoil and starts the initial extraction of the parafoil.

Steering of the parafoil is performed by winches that pull the parafoil ends down to turn the CM. NASA has developed a guidance system for the parafoil for maximum reliable performance. The system is able to autonomously fly the CM and can be overridden with manual controls if necessary.

At an altitude of 100 m the guidance system steers the CM into its final approach pattern to the landing site. At 50 m, the CM is turned into the wind to decrease the approach velocity. At an altitude of 8 m, the

Figure 24: Parafoil system.

parafoil is flared to reduce the vertical velocity to $2.4\,\mathrm{m{\cdot}s^{-1}}$ and the horizontal velocity to $16\,\mathrm{m{\cdot}s^{-1}}$. The flare is performed by pulling both ends of the parafoil down, increasing drag and decreasing velocity. The winches pull 0.75 m of control line per second to produce the optimal flare and maximize velocity reduction. Since the CM body is designed to experience high g loads on takeoff, it is cleared to perform a skid landing at the low velocities provided by the parafoil system. The skid landing was cleared for safety by NASA.

Edwards Air Force Base in California was chosen as the landing zone for the CM. The dried lake beds surrounding the base provide soft ground clear of debris, perfect for a skid landing. Drop tests show the average skid distance to be about 30 m. Upon landing the crew will be recovered by Air Force Personnel and the CM will be returned to the base via helicopter at most a few kilometers away. Water landings will be used for any emergency conditions preventing a landing at Edwards Air Force Base.

7 Exceptions to Technical Requirements

Team LETO made an extensive effort to comply with all the requirements.

All the technical requirements provided in the RFP and VSE were studied in detail by TEAM LETO to ensure that it complies with all of them.

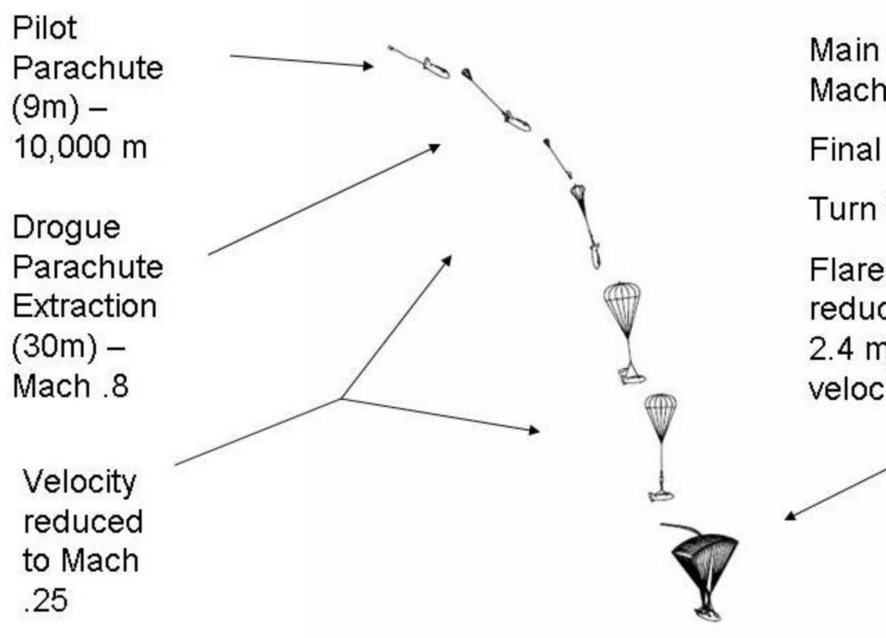


Figure 25: Project Artemis Landing Sequence.

8 Manufacturability

Project Artemis' highly modular design makes it very easily manufacturable. This helps to reduce cost, and eases the burdens of testing.

The spacecraft is divided into three main modules. These are the CM, SM, and LM. Each module is designed to be manufactured separately from the others. This eases the manufacturing process considerably. After each module is built, they are able to integrate together seamlessly. Modularity is also included at the subsystem level. Subsystems that are manufactured separately reduce cost and eases testing. This also means smaller companies can be employed for the design and manufacturing of subsystems. Each module is designed to be manufactured using technologies currently employed in the space design industry. This prevents any new manufacturing technologies from having to be developed. Doing this helps to ensure that Project Artemis will be flight worthy by 2018, as indicated by Team LETO's master schedule.

9 Mass & Power Summary

The WBS emphasizes the low mass that the Artemis Project effectively lowers the total cost of the mission. The power budget shows the solar panels were sized to the subsystem requirements.

The mass breakdown is dominated by structures, life support and propulsion. The structure system has the highest mass due to the high loads experienced by the spacecraft during the lunar operations phase. Water and water tank mass are the biggest drivers from life support. The power budget increased by 50% from the Apollo mission, but is on par with Team LETO's preliminary estimations. The largest driver for power is by far the life support system. The margins placed upon the spacecraft were 30% for mass and 20% for Power.

10 Recurring & Non-Recurring Costs

Project Artemis used NASA's Advanced Mission Cost Model to calculate the development and production cost. A total cost of $68.9 billion, which includes manufacturing and production cost for 5 vehicles, was calculated. This cost includes only the development of command, service and lunar module. The total recurring cost that would occur with the production of each set of modules was currently best estimated to be $27.56 billion. Sixty percent of the cost mentioned includes one time only non-recurring developing cost for lunar and command module.

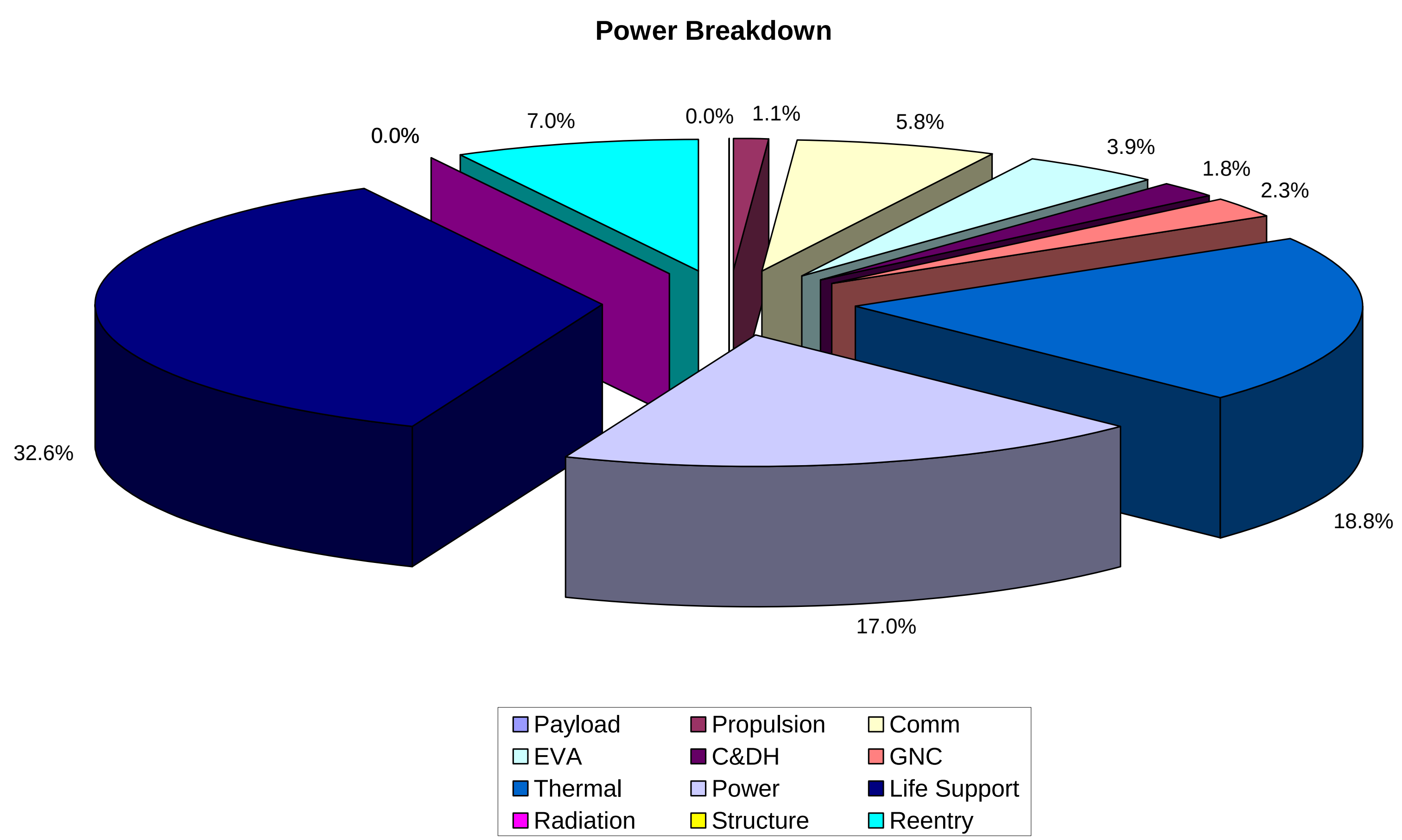
The reasearch and development cost of the CaLV was calculated to be approximately $12.6 billion with a cost per launch of $417 million. This makes the cost of the entire program which includes 10 launches and 5 vehicles over a period of 10 year to be $90 billion (FY04).

Mass Breakdown	Quantity	Mass
1 Payload	—	**820.0**
1.2 Return to Earth Mass	—	**100.0**
1.3 To Moon Mass	—	**400.0**
1.4 Passengers	4	**320.0**
2 Spacecraft Subsystems	—	**15 903.7**
2.1 Propulsion	—	**1748.3**
2.1.1 Mass of Thrusters	28	56.0
2.1.2 Mass of Engines	—	498.0
2.1.2.1 SM	1	249.0
2.1.2.2 LM	1	249.0
2.1.3 Pressurant Mass	—	5.6
2.1.4 Pressurant Tank Mass	—	1.9
2.1.5 Propellant Tank Mass	—	1062.9
2.1.5.1 RCS	—	14.0
2.1.5.2 SM	—	465.1
2.1.5.3 LM	—	583.8
2.1.6 Lines, Valves, Fittings, Regulators, etc.	—	123.9
2.2 EVA	—	**510.0**
2.2.1 Airlock	1	275.0
2.2.2 Suits	5	225.0
2.2.3 Dirt Mitigation	5	10.0
2.3 Communications	—	**22.4**
2.3.1 Transponders	2	7.6
2.3.2 Filters	2	3.0
2.3.3 Antennas	—	11.8
2.3.3.1 Hemi	2	7.9
2.3.3.2 Parabolic	1	3.9
2.4 Command and Data Handling	—	**36.8**
2.4.1 Computer	4	10.8
2.4.2 Keyboard	4	2.0
2.4.3 Monitor	4	24.0
2.5 Guidance and Navigation Control	2	**56.1**
2.5.1 IMU	1	3.3
2.5.2 Star Sensors	2	4.8
2.5.3 Momentum Wheels	2	20.0
Units	—	kg

Mass Breakdown	Quantity	Mass
2.6 Thermal	—	**1586.5**
2.6.1 MLI	—	795.0
2.6.1.1 SM	—	414.0
2.6.1.2 LM	—	381.0
2.6.2 Radiators	—	473.9
2.6.3 Coolant Loops	—	172.7
2.6.4 Heaters	—	44.1
2.6.5 Heat Exchangers	—	21.5
2.6.6 Electronics	—	79.3
2.7 Power	—	**919.0**
2.7.1 Solar Arrays	2	35.3
2.7.2 Li-Ion Batteries	4	324.0
2.7.3 PMAD	—	141.6
2.7.4 Wiring	—	418.1
2.8 Life Support	—	**2545.6**
2.8.1 Water + Tank Mass	—	1382.8
2.8.2 Air + Tank Mass	—	228.0
2.8.3 Waste Tank Mass	—	2.9
2.8.4 Atmospheric Control	—	140.5
2.8.5 Food	—	240.8
2.8.6 Personal Supplies	—	535.7
2.8.7 Plumbing	—	15.0
2.8 Radiation	—	**435**
2.9 Structure and Mechanisms	—	**6330.0**
2.9.1 CM	—	1500.0
2.9.2 SM	—	1830.0
2.9.3 LM	—	3000.0
2.10 Reentry	—	**1713.9**
2.10.1 TPS	—	1159.4
2.10.1.1 RCC	—	45.0
2.10.1.2 ARMOR	—	676.6
2.10.1.3 AFRSI	—	282.9
2.10.1.4 FRSI	—	155.0
2.10.2 Parafoil	—	554.5
Units	—	kg

Mass Breakdown	Quantity	Mass
3 Margin	30% —	**5017.1**
4 Total Spacecraft Dry Weight	— —	**15 903.7**
5 Dry Mass + Payload	— —	**16 723.7**
6 Dry Mass + Payload + Margin	— —	**21 740.9**
7 S/C Propellant Mass	— —	**50 008.7**
8 Abort System	— —	**4170.0**
9 Loaded Mass	— —	**71 749.6**
10 Boosted Mass	—	**75 919.6**
Units	—	kg

Table 21: Project Artemis: Mass Breakdown


Power Breakdown
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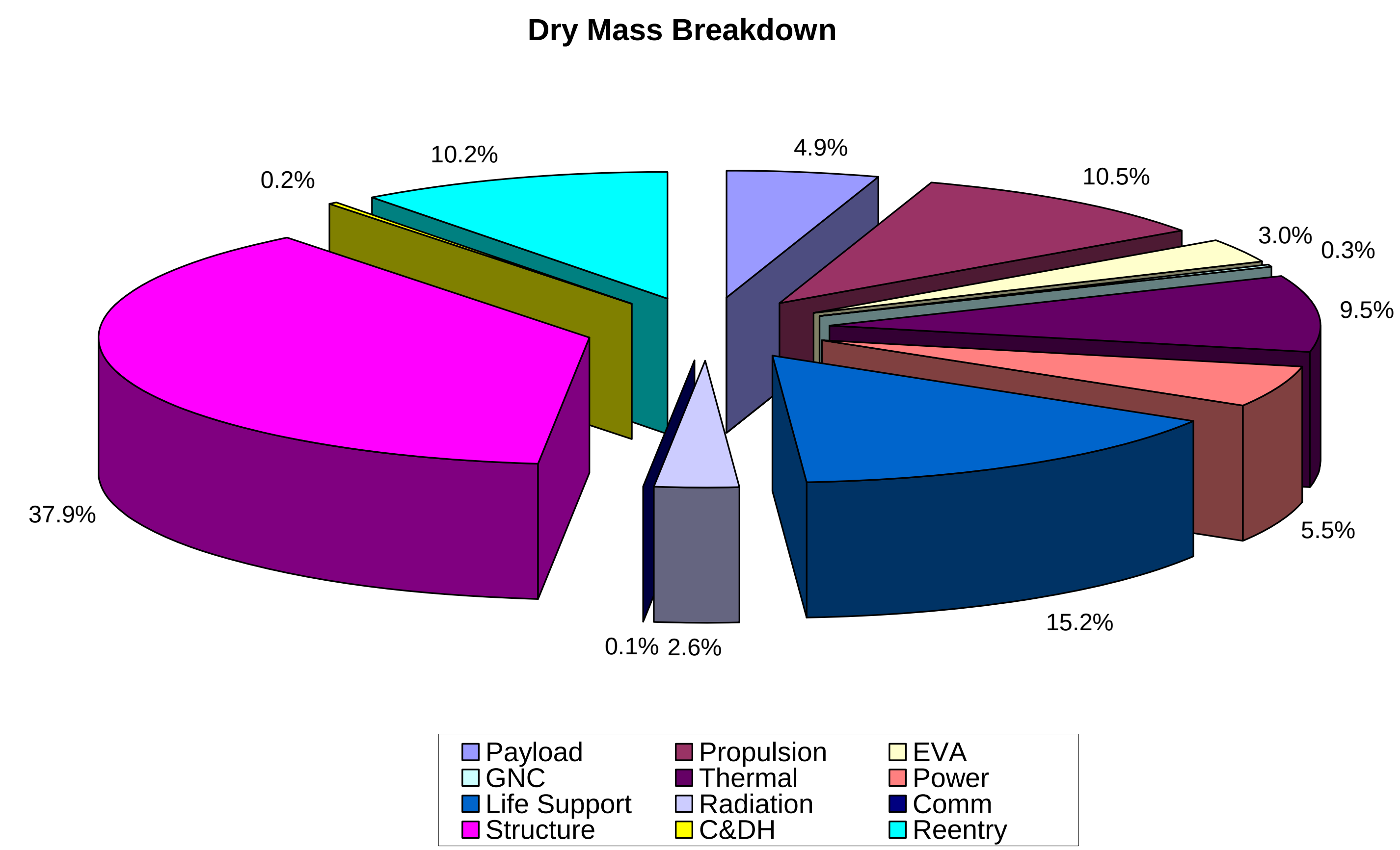
7.0%
0.0%
1.1%
5.8%
3.9%
1.8%
2.3%
32.6%
18.8%
17.0%
Payload
Propulsion
Comm
EVA
C&DH
GNC
Thermal
Power
Life Support
Radiation
Structure
Reentry


Dry Mass Breakdown
0.2%
10.2%
4.9%
10.5%
3.0%
0.3%
9.5%
37.9%
5.5%
15.2%
0.1%
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Payload
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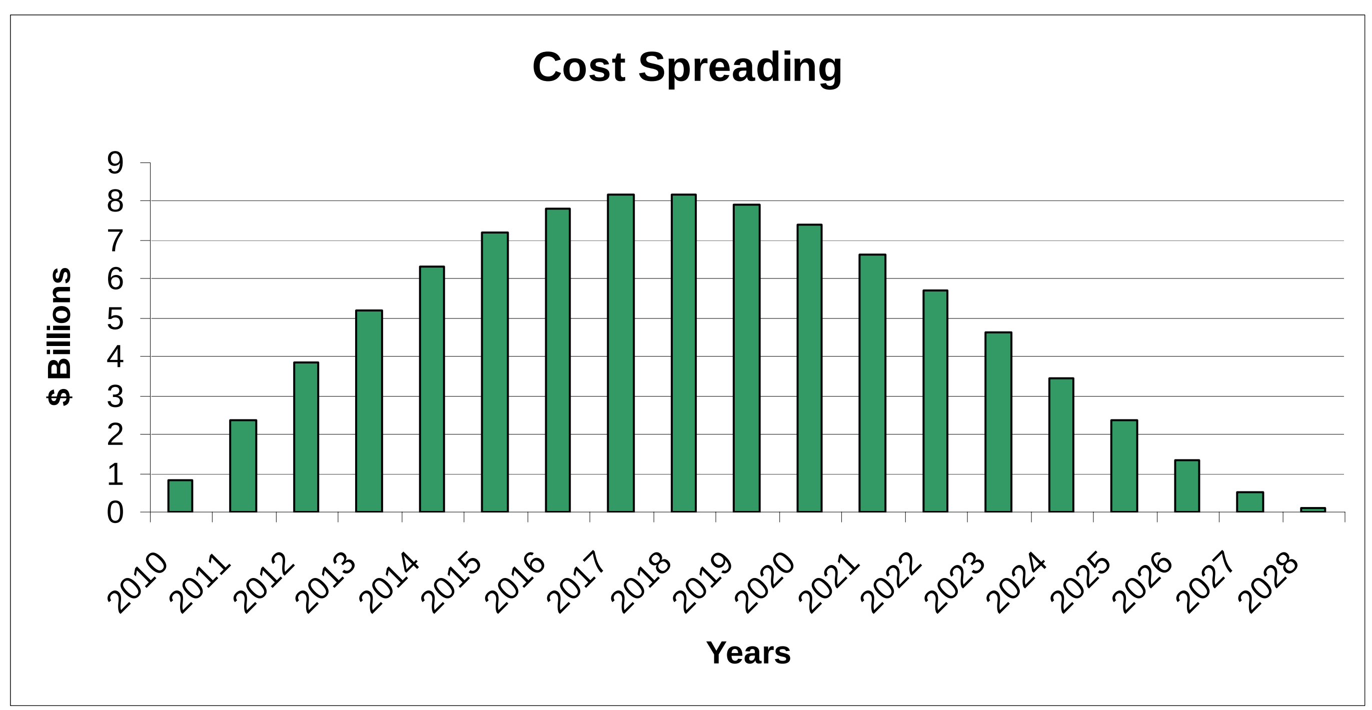
Thermal
Power
Life Support
Radiation
Comm
Structure
C&DH
Reentry


Cost Spreading
$ Billions
0
1
2
3
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6
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8
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2010
2011
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2018
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2020
2021
2022
2023
2024
2025
2026
2027
2028
Years

11 Master Schedule

Project Artemis utilizes the broad experience of the engineers at Space-Works Engineering, Inc. to ensure that the project stays on schedule during its development and production phase.

The development of the Command and the service module begins in the fourth quarter of the 2008 and continues until 2012. The first risk reduction flight will take place in the third quarter of 2013 and the second in the fourth quarter of 2014. This flight are planned to identify the potential risks before going into a full production.

Similarly the schedule for Lunar Module, the Entry Descent and Landing and the ground station control schedule is shown in detail. Ample margins have been applied in the schedule to ensure that the vehicle is ready to launch befor 2020.

12 Concluding Remarks

The Artemis Project is the best system for a safe and reliable flight back to the moon.

Team LETO has responded to the AIAA RFP by creating a space vehicle that meets all the requirements set forth. The Artemis Project also follows "The Vision for Space Exploration". The architecture is an EOR-Direct architecture, consisting of two launches with a boosted mass of 72 mT. The winged body vehicle consists of three different modules, the Command Module, the Service Module and the Lunar Module. The modularity allows the crew to discard mass as the spacecraft transits through its different phases. The vehicle can hold a crew of four and the required cargo mass.

Team LETO used Morphological Matrix, Design Structure Matrix and Analytical Heirarchy Process to determine the optimized solution for the overall system. The Spacecraft uses RS-72 a NTO/MMH bipropellant engines for lunar landing and take off and Aerojet 445 as RCS thrusters. The required power of 6000 W is supplied by 17.1 sq. m of Ultra-Triple Junction Gallium Arsenide

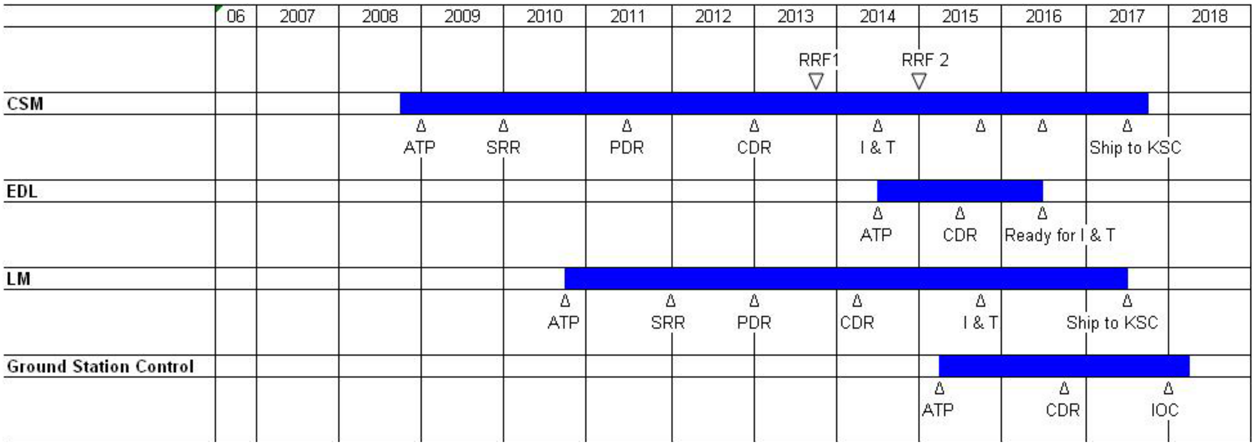

Figure 26: Integrated Master Schedule.

solar pannels. The temperature within the vehicle is maintained around 25 C by using both active and passive systems involving MLI blankets and TPS materials.

Team LETO recognized that due to the unique concept ample margins were put forth to mitigate concerns over a realistic solution. All subsystems were optimized for the safest and most reliable solution s. The TRL levels of the systems are generally high, except where Team LETO felt safety needed to be improved greatly with a new system. The schedule was created with the VSE in mind and the cost was kept at a minimum.

As the best solution for a lunar transportation system, the Artemis Project keeps the astronauts safe and progresses the nations vision to return to the moon and beyond.

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